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INTRODUCTION

Low-thrust propulsion systems that may be suitable for operations from satellite orbits will be discussed. High-thrust propulsion systems capable of launching sizable payloads into satellite orbits are considered herein only for comparison purposes. Some of the uses for propulsion systems once satellites have been established are as follows:

(1) Increasing lifetime of low-altitude satellite

(2) Controlling and altering satellite orbits

(3) Lunar and interplanetary exploration

Maintaining satellites in relatively low orbits, say of the order of 100 miles in altitude, may be desirable for observation of the Earth or as a missile-launching platform. At such altitude lifetime would be short unless a small, long-duration thrust is provided to overcome the drag.

Altering or controlling satellite orbits at higher altitudes may be desired to correct perturbations or launching errors or to reorient the satellite orbit into a more favorable location for launching vehicles to other bodies in the solar system. Again, a very small but continuous thrust would be adequate unless a rapid change in orbit is desired.

Lunar and interplanetary expeditions could range in magnitude from a small instrumented one-way vehicle to a fully manned expedition capable of landing on and exploring another planet. Again, since the

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*This report is an unclassified version of material presented originally at the NACA Flight Propulsion Conference, Nov. 22, 1957. More detailed reports on some portions of the material contained herein have since been published in references 1, 2, and 3.

journey will start from an established satellite orbit, a small but long-duration thrust will suffice to move the vehicle out of its original orbit and eventually out of the Earth's gravitational field.

In addition to these propulsion applications, auxiliary electric powerplants will be needed aboard the vehicles to operate instruments and to control the environment in manned vehicles.

A variety of propulsion systems might be suitable for these purposes. Chemical and nuclear rockets are capable of undertaking all these missions. The recombination or solar ramjet might be used to sustain a satellite in an orbit at relatively low altitude. Various electric systems are possible for all the missions that have been mentioned.

Most of this discussion will be devoted to various types of electric propulsion systems chiefly because there are so many possibilities and because some of them look quite promising. The possibilities for electric propulsion systems are listed in the following table:

Basic energy sources	Electric power generators	Thrust generators
Chemicals	Chemical batteries	Electric-arc chambers
Radioisotopes	Radioisotope batteries	Ion accelerators
Solar radiation	Thermopiles	Plasma accelerators
Nuclear fission	Solar batteries	Photon accelerators
Nuclear fusion	Turboelectric generators	
	Induction from moving plasma	

These basic energy sources, electric power sources, and thrust generators can be combined in a variety of ways. Most of the feasible combinations will be discussed later; but, since there are so many possible systems, it is desirable to discuss their common characteristics first.

GENERAL CHARACTERISTICS OF ELECTRIC PROPULSION SYSTEMS

The two most significant common characteristics of electric propulsion systems are:

(1) Higher specific impulse (lower propellant weight):

$$\text{Propellant weight} = \frac{(\text{Thrust})(\text{Propulsion time})}{\text{Specific impulse}} = \frac{F\tau}{I} \tag{1}$$

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(2) Higher powerplant weight:

$$\frac{\text{Powerplant weight, lb}}{\text{Jet power, kw}} = \text{Specific weight, } \alpha \tag{2}$$

$$\text{Jet power} = \frac{FI}{45.8} \tag{3}$$

Hence,

$$\text{Powerplant + Propellant weight} = F\left(\frac{\alpha I}{45.8} + \frac{\tau}{I}\right) \tag{4}$$

Electric propulsion systems can achieve higher specific impulse than chemical or nuclear rockets. Specific impulse is the velocity of the ejected particles divided by g; and ions, for example, can be accelerated to almost any desired velocity by sufficient voltage. Therefore, a given mission can be achieved with less propellant weight with an electric system than with chemical or nuclear rockets. The propellant weight is defined in equation (1). It might be assumed that the highest possible specific impulse is desired in order to reduce propellant consumption. That this is definitely not the case, however, may be seen when the second characteristic (i.e., the higher powerplant weight of electric systems) is considered.

As a figure of merit for the powerplant weight, the powerplant weight divided by the jet power is used (eq. (2)). This is somewhat different from the usual definition of specific weight, which is generally defined as the powerplant weight per unit thrust. For electric systems, however, the weight depends on the electric power produced; therefore, weight per unit power is a much more convenient definition. This specific weight is denoted by α (lb/kw). If the electric power were converted into jet power with 100-percent efficiency, α would be the same as the usual definition of specific weight for auxiliary electric powerplants (i.e., weight/unit electric power produced). This α, then, takes into account the additional inefficiency of conversion of electric power into jet power.

The jet power is proportional to thrust times specific impulse (eq. (3)). This definition is convenient for the purposes of this paper because it shows that, for a given required thrust, the jet power increases directly with the specific impulse. This means that, if there is a fixed weight per kilowatt, the total powerplant system will increase in weight as the specific impulse is increased. Adding the propellant weight and the powerplant weight equations gives equation (4), in which the powerplant term increases with specific impulse and the propellant term decreases. Thus, indefinitely high specific impulse is not desirable because the powerplant weight could become too high even though the propellant weight decreased almost to zero. In fact, an optimum specific impulse would be expected that minimizes the total powerplant plus propellant weight.

With equation (4) for powerplant plus propellant weight, the following equation can be written for the payload weight ratio:

$$\frac{W_{pay}}{W_0} = 0.95 - \frac{F}{W_0}\left(\frac{\alpha I}{45.8} + \frac{\tau}{I}\right) \tag{5}$$

where W_0 is the initial gross weight and W_{pay} is the payload weight. The factor 0.95 is due to an allowance of 5 percent for structure and miscellaneous weight. The exact value of this factor is not as important for electric systems as for chemical or nuclear rockets because the powerplant weighs much more than the structure. Since the powerplant must be carried throughout the mission, not much benefit can be expected from staging operations.

If the specific powerplant weight α and the propulsion time τ are temporarily assumed to be independent of specific impulse, the following equation for optimum specific impulse results:

$$I_{opt} = 1990\sqrt{\frac{\tau_{days}}{\alpha}} \tag{6}$$

This value minimizes the powerplant plus propellant weight and therefore maximizes the payload weight ratio. With this optimum specific impulse, the following equation for the maximum payload ratio is obtained for the electric propulsion systems:

$$\left(\frac{W_{pay}}{W_0}\right)_{max} = 0.95 - 87\,\frac{F}{W_0}\sqrt{\alpha\tau_{days}} \tag{7}$$

Maximum payload ratio is a function only of initial thrust to total weight ratio, the specific powerplant weight α, and the required propulsion time τ. An interesting point is that, for this optimum specific impulse, the propellant weight is equal to the powerplant weight.

The assumption that α and τ are independent of specific impulse is not generally valid, but it is shown later that other assumptions and more accurate analyses give about the same minimum powerplant plus propellant weight.

In order to determine the payload ratio for various missions, three parameters must be determined: the initial thrust-weight ratio, the propulsion times required to accomplish the mission, and the specific weight

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of the powerplant. The specific weights attainable with various propulsion systems will be considered later, but first the required values of thrust-weight ratio and propulsion times for the four missions mentioned in the INTRODUCTION should be indicated.

For the satellite-sustainer mission, thrust-weight ratios of the order of 10^{-5} or 10^{-6} are adequate for overcoming the drag at altitudes down to about 100 miles. The propulsion time, of course, depends on the purpose of the satellite and is limited only by the requirement that it cannot be so large that the payload becomes zero. For the orbit-control application, values of F/W_0 of the order of 10^{-5} or 10^{-6} are again adequate if fast maneuvers are unnecessary. The propulsion time again depends on the intended application.

For lunar and interplanetary voyages the required propulsion time is found by integrating the equation of motion of a vehicle propelled by a constant thrust in a gravitational field. Calculations were made for several values of the thrust-weight ratio F/W_0. A typical result is shown in figure 1. The trajectory is that followed by a vehicle propelled by a constant thrust, starting from a satellite orbit near the Earth. The initial thrust-weight ratio is 10^{-4}. The vehicle follows a spiral path with very gradual increase in distance from the Earth at first. As the gravitational field becomes weaker, the speed of recession from the Earth increases. After about 80 days, the orbit of the Moon is approached. A high-thrust rocket would reach the Moon's orbit in about 3 days. Thus, thrust-weight ratios of the order of 10^{-4} or less are not desirable for manned journeys to the Moon. If the thrust-weight ratio is increased to 10^{-3}, the Moon is reached in the more satisfactory time of about 8 days, but the required powerplant weight for $F/W_0 = 10^{-3}$ is considerably lower than appears possible with the electric propulsion systems that now appear feasible.

If the thrust corresponding to $F/W_0 = 10^{-4}$ is applied for about 47 more days (about 127 days in all), the vehicle acquires enough energy to follow the least-energy transfer ellipse from Earth's orbit to Mar's orbit. This trajectory is shown in figure 2. Also indicated in this figure are the times required to accomplish each part of the journey with an initial F/W_0 of 10^{-4} and with a high-thrust rocket. Because of the long wait required at Mars before the Earth and Mars are in a favorable position for the return trip, the total time required for the minimum-energy round trip with low thrust is comparable with that required with impulse rockets. The time difference is due entirely to the larger time required by the low-thrust device to spiral out of, and back into, a satellite orbit near the Earth's surface. (The round-trip time for the low F/W_0 is actually somewhat less than that shown (perhaps 1150 days instead of 1205) because the mass of the vehicle is less when it returns to the

Earth than when it left. This mass reduction, due to consumption of propellant and subsistence supplies, was ignored in the time computations.)

When the propulsion time for a mission has been calculated, the payload ratio available for the mission can be obtained from equation (7) for suitable values of F/W_0 and α. The equation is plotted in figure 3. For the low thrust-weight ratios required for satellite control or satellite-sustainer missions (of the order of 10^{-5} or 10^{-6}), the allowable powerplant specific weights are quite large, even for very long sustaining times. For the propulsion time required for the round-trip Mars journey, however (indicated by the circle points on each curve), specific propellant weights of the order of 20 pounds per kilowatt or less are needed in order to make the journey with $F/W_0 = 10^{-4}$ with significant payload ratio. With lower F/W_0, the required propulsion times become much too large.

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The previous discussion has assumed that values of the specific impulse near optimum can be achieved. The effect of nonoptimum specific impulse on the payload ratio for the Mars round trip is shown in figure 4 for $F/W_0 = 10^{-4}$ and for three values of α. For specific powerplant weights of the order of 10 or 20, the range of specific impulses that permit near-maximum payload ratio is rather narrow. The decrease at high specific impulse is due to the large powerplant weight required, and the decrease on the low-impulse side is, of course, due to the larger propellant weight required.

For smaller α, of the order of 1, the specific impulse is not seriously limited on the high side, but the minimum allowable value is still about 5000 seconds. The fact that the minimum value is so high may seem surprising at first since chemical rockets can accomplish the Mars mission with much lower specific impulses if enough initial weight is provided. There are two reasons for the difference. The first is that the electric systems have little staging possibility since the heavy powerplant must be carried throughout the trip. The electric propulsion system is effectively a single-stage vehicle. The second reason is that the low-thrust vehicle spends a much larger time than the impulse rocket in working out of strong gravitational fields. This means that the energy expended for a given mission is greater than for the impulse rocket, and that the "characteristic velocity" for a given mission is correspondingly higher. It is easily verified that the minimum specific impulse required to attain a given energy in a gravitational field with a single stage increases as the thrust-weight ratio decreases.

To summarize the results of this preliminary discussion: Extremely high specific impulses are not desired because the powerplant weight becomes too large. Electric propulsion systems are capable of performing the round-trip Mars mission with sizable payload if specific powerplant weight is near 20 pounds per kilowatt of jet power or less, and if specific impulses near optimum are attained. Values of F/W_0 of the order of 10^{-4} are satisfactory for the Mars mission but are too small to accomplish the Moon mission in reasonable time. Electric power required will depend on the desired size of the vehicle, but will be of the order of 200 kilowatts to 20 or 30 megawatts for interplanetary missions. For satellite sustainers, satellite orbit control, and auxiliary power, the electric power required will be of the order of a few kilowatts. The specific powerplant weight allowable is much higher for these applications because of the much lower thrust-weight ratios required (of the order of 10^{-5} to 10^{-6}).

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ELECTRIC POWER GENERATORS

The various possible combinations of the basic energy sources, electric power generators, and thrust generators listed previously will now be considered. For convenience, the feasible combinations of basic energy sources and electric power generators will be discussed first.

Chemical Batteries

The nonelectric propulsion schemes will need small, lightweight auxiliary electric power sources for instruments. The familiar sources for applications of this type are chemical batteries. The all-important factor for flight applications is the ratio of weight to power. Two disadvantages of today's chemical batteries are that these batteries are basically low voltage sources and that they can be as heavy as lead. The question is whether either of these unfavorable features can be overcome by research.

As to the voltage problem, the laws of thermodynamics indicate that the chemical cell will always be a low voltage source (i.e., less than 5 v). Higher voltages may be obtained by series grouping of cells. Therefore, the reliability of the individual cells will probably limit the total voltage of a chemical battery to a value of the order of 1000 volts.

In general, the time during which a battery must supply power determines its weight. The ratios of weight to power for a few commercial batteries at various load times are compared in figure 5. The ordinate showing this ratio has the units of pounds per watt; most of the later figures use kilowatts as a basis. The familiar lead-acid and common dry cells do not show up well in the weight comparison.

The mercury (Ruben) cell was developed during World War II for "walkie-talkie" radios because of its favorable weight and compactness.

The mercury cell has been considered for the project Vanguard satellite auxiliary power source because of its insensitivity to pressure and its temperature range (-65° to 250° F).

The silver-zinc-alkaline cell is another newcomer that has gained popularity for missile applications. It can supply near-rated ampere-hour capacity at great overloads, but it has poor temperature characteristics.

Research on fuel cells has been carried on for over 50 years, but the first commercial venture is the National Carbon H_2-O_2 cell now being used by the services for remote radar stations. It is very advantageous for long and continuous service. Unlike other cells, the electrodes of the hydrogen-oxygen cell are permanent; similarly, the liquid electrolyte needs only occasional care. These features, together with gaseous reactants that are easily fed continuously into the cell, result in a system that is uniquely suited for service over long times. For times over about 100 days, the major weight will be the hydrogen-oxygen gas containers. Theoretically, 1350 ampere-hours can be obtained from the reaction of 1 pound of gases; this is probably the limit obtainable from any chemical reaction. The H_2-O_2 fuel curve of figure 5 reflects a 4-pound container storage weight penalty for every 1/9 pound of H_2 and 8/9 pound of O_2. These gases were assumed to be stored in liquid phase for long times in containers only slightly lighter than those used today for ground storage. Obviously, some development work on this H_2-O_2 fuel cell would be required to develop a system suitable for space propulsion, but for long-time service this cell will be the best of the chemical batteries.

Other Low-Power Electric Sources

As can be seen from figure 5, the weight of chemical batteries is quite high. To achieve greater savings in weight it is desirable to carry a more compact energy source or to tap some external energy source. The first of these possibilities involves the use of nuclear energy. The specific energy (Btu/lb) is from 10^5 to 10^6 times greater for nuclear reactions than for chemical reactions. In the low power range (100 to 1000 w), the decay energy arising from radioisotopes appears attractive. By comparison, small nuclear reactors needed to produce power in this range would be heavy.

The second of the weight-saving possibilities involves external energy sources such as solar energy. Figure 6 presents a plot in pounds per watt against time in days for some proposed lightweight electrical power-plants in the 100- to 1000-watt category using a radioisotope source or

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solar energy. The radioisotope chosen, mainly because of its high specific energy and potential availability, was Po^{210}, an alpha-emitter. The H_2-O_2 fuel-cell curve from figure 5 is also shown in figure 6 for comparison.

The two horizontal lines of figure 6 represent the solar battery, the lower for the solar battery in the Sun full time, the upper for the solar battery in the Sun half time. Most of the weight difference is due to the batteries required to store electricity for the times when the latter system is not in the Sun.

The other systems shown in figure 6 are all radioisotope systems. The first system to be considered is the thermopile. In this system the heat produced by the radioisotopes is used to induce an electric current in a thermopile. Most of the weight of this system is in the thermopile itself, and, therefore, a thermopile system using solar energy would not yield substantially better weight-power ratios. A second system uses radioisotope heat to boil mercury. The mercury vapor turns a turbine driving an electric generator. The curve for this system falls well below the one for the thermopile system.

A third system, studied at the NASA Lewis Research Center, utilizes radioisotope energy in a different fashion. In this system (fig. 7) the alpha particles dissociate water into hydrogen and hydrogen peroxide in the decomposition chamber. The hydrogen, being a gas of low solubility, separates from the water stream. The hydrogen peroxide is carried by the stream to a second chamber where it passes over a catalyst and decomposes to oxygen and water. The oxygen is removed at this point. The two gases, hydrogen and oxygen, are fed into a fuel cell similar to the one developed by the National Carbon Company. In this cell the gases react to give water and electrical energy. Since the radiolytic process makes only partial use of the available energy, much of the radioisotope energy goes into heat that must be rejected from the system. The water stream is therefore passed through a radiator to remove this heat.

The radioisotope-fuel-cell system falls on about the same curve as the radioisotope - mercury-vapor system (fig. 6). However, the radioisotope-fuel-cell system shows promise of efficiency improvement through the use of semiconductor materials as intermediates in the water decomposition process (ref. 4). The possibility of a twofold or even a fourfold increase in efficiency appears good. As can be seen from figure 6, the radioisotope-fuel-cell system sensitized to twice the unsensitized efficiency gives a curve that, for periods of less than $1\frac{1}{2}$ years, falls below all except the solar battery in full sun. It must be remembered, however, that the separation of gas and liquid in the two decomposition chambers of this system has been assumed to be complete; this may not be the case, in which event the specific weight of this system would be increased.

For long periods of time the solar-energy systems appear to be best on a weight basis. For shorter times, the mercury-vapor and fuel-cell systems appear to be the best, the fuel-cell system showing potential for future improvement.

Nuclear-Electric Powerplant

Fission of uranium was considered as the energy source in an investigation of turboelectric powerplants of 500- to 20,000-kilowatt electric output. A specific configuration was selected in order that weight could be estimated for use in the propulsion study. This does not imply that either the specific configuration or the weights have been optimized but only that they are specific.

A turboelectric powerplant could be arranged as shown in figure 8(a). If the working fluid is a gas, the gas could be heated in the reactor, expanded in the turbine, cooled in the radiator, and compressed by the compressor to its initial pressure, thereby completing the cycle. In space, heat must be rejected from the radiator by thermal radiation rather than by convection because there is no air to act as a heat sink. If the fluid entering the reactor is a liquid, the liquid could be boiled by the heat addition in the reactor. In this case, the resulting vapor would be condensed in the radiator, and the compressor would be replaced by a pump.

Shielding of the reactor is required in order to protect the crew. If the cycle's working fluid becomes radioactive on passing through the reactor, the shielding problem is considerably complicated because all components of the cycle (the turbine, the radiator, and the pump) also then release radiation requiring shielding. This activation of the working fluid can be avoided by introducing an intermediate heat exchanger as shown in figure 8(b). One fluid passes through only the reactor and the intermediate heat exchanger. Another fluid is heated in the intermediate heat exchanger and used as the cycle's working fluid. In this way, the turbine, the pump, and the radiator do not become radioactive.

Since original studies showed the radiator to be very large, ways of reducing radiator size were investigated. The variation in radiator area with radiator temperature is shown in figure 9, where two classes of working fluid are compared for a single turbine-inlet temperature of 2040° F. The helium curve indicates what can be accomplished by using gases. The large compressor work penalizes gas cycles and requires a low temperature entering the radiator. As a consequence, the radiator areas per kilowatt of electrical output are large for helium. The two vapor cycles shown are comparable in radiator area, sodium being a little better because of its higher critical temperature. The attainable radiator areas within a given temperature limit are better by more than an order of magnitude for vapors than for gases. The remainder of this study therefore considers only vapor cycles. For sodium, a radiator area of 0.8 square foot per kilowatt is required for a radiator temperature of 1340° F.

For the temperature shown, sodium is superior to mercury because of the pressures involved. At 2040° F, mercury boils at 5400 pounds per square inch. At 1340° F, mercury condenses at 900 pounds per square inch. These pressures will add to the powerplant weight by requiring heavy walls. The cross section of sodium for capture of thermal neutrons is also considerably superior to that of natural mercury. This disadvantage of mercury could be largely eliminated by isotopic separation of the mercury, but sodium was chosen as the more promising working fluid for further study.

The pressures and temperatures of a sodium cycle are shown in figure 10 along with a schematic arrangement of the powerplant. Liquid sodium at 2340° F circulates through the reactor and the heat exchanger. The absence of oxygen in space will help to permit operation at these temperatures. In the heat exchanger, the cycle working fluid is heated to 2240° F. The pressure of 200 pounds per square inch is sufficient to keep the sodium a liquid even at 2340° F. This liquid sodium then enters the evaporator. At 2040° F and 70 pounds per square inch, $3\frac{1}{2}$ percent of the sodium leaves as a vapor. The sodium vapor expands in the turbine to 2.7 pounds per square inch, producing about 400 Btu's from each pound of sodium passing through the turbine. The overall cycle efficiency is 20 percent. Condensation of vapor in the radiator presents a problem in removing liquid from the tube walls that does not exist on Earth because of the gravitational field. The whole powerplant could be rotated about a longitudinal axis in order that centrifugal force could keep the condensed liquid moving along the walls of the radiator. This rotation would also provide an artificial gravity field for the crew.

The estimated weights of such a powerplant are shown in figure 11. At 20,000 kilowatts, the radiator has the dominant weight, in spite of the fact that the design was varied to minimize this weight. The generator and reactor make significant but small contributions. The miscellaneous item includes the heat exchanger, pumps, turbine, evaporator, piping, sodium, and structure, none of which individually adds much to the weight. At 20,000 kilowatts, the shield weighs about 1 pound per kilowatt. As the design value of power changes, the weight per kilowatt of most of these items remains essentially constant. The shield is an obvious exception. As the design power changes, shield weight changes slowly, with the result that its weight per kilowatt climbs steeply as power goes down.

At 20,000 kilowatts, the estimated total powerplant weight is $5\frac{1}{2}$ pounds per kilowatt. Power levels below 500 kilowatts were not considered in this investigation, but the specific weight will clearly rise rapidly as power is decreased, probably being of the order of 25 to 200 pounds per kilowatt in the power range below 50 kilowatts.

Weight estimates such as that in figure 11 must be predicted on some presumed geometric configuration. The geometry considered in the weight estimation for figure 11 is shown in figure 12. For 20,000 kilowatts, the overall length is 600 feet. The radiator dominates in terms of physical size as well as weight. The crew compartment is separated from the reactor in order to reduce the shielding requirements. In order to tie the whole device together, a $2\frac{1}{2}$-foot-diameter tube is provided that will take both tension and compression; such a structure will keep the crew compartment away from the reactor. This tube also has some strength as a beam and will supply some stiffness to the whole vehicle.

The reactor and turbine ends of the powerplant are shown in somewhat more detail in figures 13(a) and (b), respectively. The problem of shielding the crew from the reactor is simplified in outer space because of the absence of any air for scattering. For this reason shadow shielding was used for the crew compartment, the radiator, and all the machinery in the shadow of the shield. The shielding was designed without regard for the less well-known effects of cosmic radiation. A literature survey indicates that rather heavy crew shielding is required to protect against cosmic radiation. In spite of such shielding of the crew compartment, it appears that shielding of the reactor will still be required.

Evaporation of liquid into vapor in the absence of a gravity field presents a problem of separating the two phases similar to that encountered in the radiator. Location of the heat exchanger and the evaporator near the axis of rotation of the vehicle keeps low the centrifugal acceleration within these pieces of equipment. For this reason, the 130-pound-per-square-inch drop across the evaporator was exploited to produce a rapid rotary movement of the liquid sodium within the evaporator. About 1000 g's of radial acceleration are available for separation of the vapor from the liquid.

The number of turbine stages is sensitive to the allowable centrifugal stress in the turbine blades. Two stages are shown for the turbine in spite of the work requirement of 400 Btu's per pound.

Radiator weight was kept low by assuming that the radiator could be built of tubes having a wall thickness of 0.020 inch. Walls of this thickness are susceptible to damage by meteoroids. Reference 5 indicates that, on the average, such a radiator for a 20,000-kilowatt powerplant will suffer one penetration by a meteoroid each 40 days; only one in 10,000 such holes will be bigger than 1/4 inch. The radiator is segmented in order that valves in the manifolding can isolate the damage from a meteoroid until the resulting leak can be repaired. Thicker walls for the radiator should decrease the incidence of damage by meteoroids; a 0.025-inch wall thickness will increase the average time between penetrations to 100 days, and it will increase the powerplant weight by 5000 pounds, or 5 percent. Damage by meteoroids cannot be avoided with certainty because of the extreme penetration of rare particles. Other estimates of damage by meteoroids disagree by an order of magnitude (ref. 6, e.g.). If these estimates are correct, the radiator described previously would be inadequate from the point of view of meteoroid penetration and would have to be redesigned.

In summary, this hypothetical powerplant has three salient features: (1) The working fluid is a vapor; (2) the radiator is very light in construction, depending on the ability to recover from meteoroid damage; (3) the operating temperatures are fairly high. Failure to incorporate these three characteristics will result in a big increase in powerplant weight.

Solar Turboelectric System

About 100 watts of solar radiation are incident on a square foot of surface normal to the Sun at the Earth's orbit. A possible scheme for using this energy is shown in the block diagram of figure 14(a). The circuit is identical with that just described for the nuclear-electric system, except that the reactor is replaced by a very large mirror that focuses the solar energy on a heat exchanger. (The present system is proposed rather than one using thermopiles because the thermopiles make the weight much greater.)

The main problem in any solar system is the mirror. A possible arrangement is shown in figure 14(b). The mirror is a large polyester balloon, as proposed in reference 7. Half is transparent and half is silvered. The heat is focused on a heat exchanger that extends along the axis from the mirror to a point halfway from the axis to the mirror. The remainder of the cycle is assumed to be exactly the same as in the nuclear-electric system. The rotating machinery is placed inside the balloon to limit the length of hot lines and also for stability. The crew compartment would also be there. Thrust chambers would be outside, as would controls for aiming them. In addition, the sphere must be separately controlled so that the mirror always faces the Sun.

To get 20 megawatts of electric power, a balloon diameter of about 1260 feet is needed. Such a balloon, made of 1-mil-thick Mylar, weighs about 36,000 pounds. To obtain a total weight estimate, the same weights are used as for the nuclear-electric system, making allowance for controls. Then the total weight for a 20-megawatt electric power output is about 110,000 pounds, which is virtually the same as for the corresponding nuclear system. If, for some reason, a rigid mirror, not a balloon, is desired, the weight will be very much greater. If a number of small balloons replaced the single large one, a lesser weight penalty would be involved.

The main advantage of this scheme over the nuclear-electric system is that no shield is needed. Hence the equipment is readily accessible. This also means that at lower powers a weight advantage should occur because most of the components will scale more or less linearly. For example, it is estimated that a 200-kilowatt electric power system would weigh on the order of 1500 pounds.

On the other hand, there are serious difficulties. The power available varies as the square of the distance from the Sun. The power at Mars is about 40 percent of that at Earth. Another problem is that near a planet the vehicle may be shielded from the Sun's rays. Then no power is delivered, and large storage facilities may be needed. Finally, the size makes meteor damage more likely. Although repairs are simple and very little gas is required to inflate the balloon, this problem might well make its use impossible. In such a case, a rigid and, unfortunately, heavy mirror would be required.

The use of this system for a satellite sustainer is improbable, as the balloon probably cannot overcome its own drag at altitudes of less than about 300 miles. It might, however, be used at higher altitudes for orbit control or auxiliary power. For such applications, about 3 kilowatts of electric power might be obtained for about 300 pounds weight, very little of which is in the 15-foot mirror.

Nuclear Fusion

Both fission and solar power have been considered. Perhaps fusion will someday have a place in this type of application.

There is little to gain by using fusion energy as a heat source for a thermodynamic cycle with a working fluid. Such a system would be very similar to the fission reactor system that uses a sodium-vapor cycle and involves heat exchangers, radiators, and turbines. The thermonuclear machine would merely replace the reactor of the fission system, and the reactor is only a small portion of the weight of that system. Therefore, there is no particular advantage to a thermonuclear machine used strictly as a means of heating a working fluid.

The direct production of both electricity and thrust should be possible with a thermonuclear machine and probably could be realized in the future. The weight of such a system is difficult to estimate. Although thermonuclear theory is not far advanced, a few observations can be made.

All current thermonuclear machines of interest utilize one of two types of magnetic fields for containing the high-temperature plasma, a magnetic field produced by external field coils or a magnetic field induced by high currents in the plasma itself. The principal weight associated with the first is the weight of the field windings. At present, large-volume magnetic fields can be wound with field strengths of the order of 50 kilogausses; and with fields of this strength the weight of the thermonuclear machines would be much greater than those of the fission or solar systems discussed previously. Production of higher field strengths is being investigated; and, if field strengths of the order of 200 kilogausses or higher can be obtained, some of the machines using externally wound fields might become of interest.

The principal weight associated with the machines that rely on a current-induced magnetic field is the weight of the condenser bank used to produce the high plasma current required. Recent advances in the use of mixtures containing barium titanate as a dielectric material offer the hope that this weight can be reduced to manageable proportions. A rough estimate was made of the weight per jet kilowatt of a machine using a current-induced magnetic field for confinement. A stabilized pinch machine was considered, and weights were estimated for its various components such as the main condenser bank, the stabilizing field condenser bank and coil, the preheating or "collapse" field condenser bank, the vacuum system, the neutron shield, and the cooling system. These estimates are very uncertain; but, if such a system can be made to work, the weight per jet kilowatt might be of the order of 3 pounds, or the thrust-weight ratio of the order of 8×10^{-4} for powerplants of the larger sizes being considered.

Comparison of Electric Power Generators

Most of the feasible combinations of basic energy sources and electric power generators have been discussed. A comparison of the more promising ones is shown in figure 15, in which the estimated weight of several systems is plotted against the electric power output. The bars near the top of this figure show the ranges of power required for the missions being considered. The nuclear turboelectric system without shielding and the solar turboelectric system in the Sun full time are comparable in weight throughout the power spectrum. For auxiliary power and the satellite sustainer and control applications, a number of systems are competitive. The weight of the solar battery and solar turboelectric systems will depend greatly on the penalty in weight necessary for part-time-in-Sun operation. For propulsion applications, this penalty need never be more than twice the weight for full time in Sun, since the mission of interest can be performed by applying proportionately greater thrust during the time that the vehicle is in the Sun.

For the higher power levels required for lunar and Mars missions, the only systems that remain competitive are the nuclear turboelectric, solar turboelectric, and the fusion-electric generator. With increasing power, the shielding weight becomes a less significant percentage, and both the nuclear turboelectric and the solar turboelectric systems approach a linear variation of weight with power. The straight line at higher powers corresponds to a slope of about 5 pounds per kilowatt. If the electric power could be converted into jet power with high efficiency and with little additional weight in thrust-generator apparatus, this value would represent approximately the specific powerplant weight α and would be a very satisfactory value for the Mars mission. It is shown later that the required additional weight of the thrust generators will be moderate for ion or plasma accelerators, so that the value of α will depend principally on the efficiency of conversion of electric power to jet power.

The single weight estimate for the fusion-electric system indicates a weight of about 3 pounds per kilowatt at 20 megawatts. The curve of weight against power output will probably be more nearly horizontal for this system than for the fission turboelectric system; consequently, it appears that the fusion-electric system might be applicable chiefly to large-scale expeditions to Mars and beyond.

THRUST GENERATORS

Electric-Arc Chambers

One method of generating thrust with electric power is to heat a propellant with an electric-arc discharge. This method is illustrated in figure 16. The arc chamber is similar to a rocket motor, the principal difference being that the propellant is heated electrically instead of chemically. An arc is struck from the anode to the nozzle walls, which serve as the negative electrode. The propellant can be passed along the combustion chamber and the anode to provide regenerative cooling. The propellant then passes through the electric arc, where it is heated, and expands through the nozzle. If the power level is so high that the propellant does not provide sufficient cooling, another cooling circuit must be provided, and the excess heat must be rejected through radiation.

There are three basic limitations to the specific impulse attainable with an electric-arc chamber: (1) electrode consumption rate, (2) local heat-transfer rate at the throat, and (3) overall cooling rate. An estimate based on electrode consumption rate for an uncooled graphite anode indicates that the maximum specific impulse for this case may be limited to about 1500 seconds. This limit arises when all electric power goes into vaporizing the electrode and none goes into the propellant.

If a method is found for overcoming the limitation due to electrode consumption, the cooling requirements impose further limitations. The throat heat-transfer rate becomes severe at high chamber pressures but can be brought down to reasonable values by decreasing the pressure. However, going to low pressures increases the fraction of the total enthalpy that must be removed to maintain a given maximum allowable surface temperature throughout the arc chamber and nozzle.

The effect of overall cooling requirement on specific impulse is shown in figure 17 for arc chambers designed to produce 1 pound and 100,000 pounds of thrust. These curves are for a 10-atmosphere chamber pressure and for a nozzle that produces 75 percent of the maximum specific impulse. If a larger nozzle or a lower pressure is used, the limitation on maximum specific impulse becomes more severe. If a higher pressure is used, the throat heat-transfer rate becomes excessive. Figure 17 shows that, as electric power is increased, specific impulse increases almost linearly first but then reaches a maximum value. At this maximum, further increases in power must be removed by cooling to maintain the allowable surface temperature. The maximum specific impulse for the low-thrust nozzle is about 2000 seconds and for the high-thrust nozzle about 4000 seconds. This specific impulse is not particularly high. Figure 4 showed that, for a thrust-weight ratio of 10^{-4}, at least 5000 seconds are needed to undertake the Mars round-trip mission. The electric-arc-chamber propulsion system is therefore marginal for this mission but may be useful for lower energy missions.

A possibility not yet mentioned is that, at higher temperatures, for which the ionization becomes more significant, magnetic fields might be used to keep the hot propellant away from the surfaces. The cooling limitation on specific impulse might thereby be alleviated. This possibility has not been examined in detail, but additional weight in field coils and additional electric power would certainly be required. Furthermore, other electric thrust generators are capable of achieving higher specific impulse and comparable thrust-weight ratios without going through the propellant heating cycle. It may therefore be concluded that the electric-arc chamber is not a very promising method of generating thrust from electric power for application to round-trip interplanetary missions, but that the method may be useful for more moderate missions which require lower specific impulse.

Thrust from Ion Acceleration

The arc system suffers from overheating. A scheme that does not have a real heat difficulty is one in which ions are generated and then accelerated electrostatically. Figure 18 shows the parts of such a system. The propellant is ionized and then accelerated electrostatically and finally exhausted to space. The accelerator will require about 20,000 volts of direct current. The power-generating systems discussed earlier were designed on the basis of low-voltage alternating current. However, if a high-voltage alternating-current generator is used and rectifiers are added to get direct current, only a minor weight penalty is incurred.

The items shown in figure 18 will not give any thrust at all. If only positive ions are emitted, a space charge will immediately build up outside the ship. This effective decelerating potential will immediately stop all flow. Hence, electrons should be emitted at the same rate to neutralize the charge. Fortunately, this is not difficult. The main design problem associated with this space charge is that charge neutralization must occur in a very short distance - of the order of a fraction of an inch - if reasonable current densities are desired.

A second design problem is that the ionization chamber must be simple and able to operate for long periods. In addition, it should ionize all the propellant. What is not ionized will not be accelerated to very large velocity and so will be wasted.

One especially simple ionization scheme has been suggested. The alkali metals (cesium, rubidium, and potassium) have low first-ionization potentials - about 4 electron volts. On the other hand, the work function of heated platinum or tungsten is larger than this. Experiments conducted about 30 years ago (refs. 8 and 9) showed that, when such an alkali vapor is passed over a suitably heated tungsten plate, the atom is adsorbed and reemitted as a positive ion. If the plates are at about 1800° F, the probability of ionization is virtually 100 percent (ref. 10).

Figure 19 shows a system using this ionization method. The arrangement is due to Stuhlinger (ref. 11), as is the idea of using this method of generating ions. A vapor of cesium is admitted and passed over a series of heated plates that are maintained at a small potential difference. Beyond the plates, a large potential difference is maintained to accelerate the ions to the desired final velocity. They are emitted to space as are electrons to neutralize the space charge.

It is desirable to have as large a current density in the jet as possible because the total current may be as much as 500 amperes or more. One severe limitation is the external space charge:

$$\text{Limiting current density} \propto \frac{1}{(\text{Distance})^2} \sqrt{\frac{\text{Charge}}{\text{Mass}} (\text{Voltage})^3} \qquad (8)$$

The allowable current density varies with the charge, mass, and voltage through which the ions have been accelerated; but, most important, it varies inversely as the square of the distance to neutralization. For example, for cesium at an impulse of 15,000 seconds, the required accelerating potential is 10,000 volts. If neutralization takes place in 1 inch, a current density of only 1 ampere per square foot is allowed. This can lead to exhaust areas of the order of 500 square feet.

Now if the total current required is known in terms of the thrust and impulse, the required jet area can be written as follows:

$$\text{Jet area} \propto \frac{\text{Thrust}}{(\text{Specific impulse})^4} \left(\frac{\text{Charge} \times \text{Distance}}{\text{Mass}}\right)^2 \qquad (9)$$

What is wanted is heavy ions, singly charged. This is fortunate, because the single charge is the most easily obtained. Also desired are rapid neutralization and high specific impulse.

There are two main reasons for wanting low jet area. One is concerned with thermal radiation loss, which could be exorbitant. The other is that the weight will increase with the area. If it is arbitrarily assumed that the weight is proportional to this area, some weight estimates can be made for a nuclear-electric system flying to Mars. Table I shows the results for a Mars mission. The first two cases correspond to 4 pounds per kilowatt at a 15,000-second impulse, with variation as in the area equation (9). Efficiencies of 40 and 80 percent are assumed. An optimum impulse of about 20,000 seconds is found, and the propulsion-system weights are on the order of 6000 to 8000 pounds per pound of thrust. If the ion source weighs nothing at all, slightly smaller results follow. If the current density is not governing, the area equation will not set the weight. Then it might vary linearly with impulse. Such a case is shown in the last line of table I. Two general results follow from these estimations. Regardless of the assumptions concerning the weight of the ion accelerator, the optimum impulse is in the range of 14,000 to 23,000 seconds for all cases considered. The required accelerating potential will therefore be under about 35,000 volts. The weights all show that a thrust to gross vehicle weight ratio of 10^{-3} is not possible but that 10^{-4} is attainable. This means that the Mars mission can be accomplished in reasonable time.

If the jet can be neutralized very rapidly (say, in 1/10 in.), high current densities are possible. Then the possible rate at which ions can be generated limits the size. In this case the contact method of ion generation may not be satisfactory. However, if electron bombardment is used, very high current densities can be obtained (on the order of several hundred amp/sq ft). However, this method has a certain complexity and may have lower ionization efficiency. Considerable study of such systems is in order.

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Plasma Accelerators

As a result of large space charges built up at the accelerator exit, the cross-sectional area of the ion accelerator is large. If ions and electrons were both accelerated in the same direction and charge neutrality preserved, much higher particle densities and much smaller accelerators would be possible. An ionized gas in which charge neutrality is preserved is called a plasma, and the associated accelerator might be called a plasma accelerator. Since electric fields tend to accelerate oppositely charged particles in opposite directions, magnetic fields must be used to accelerate plasmas.

One idea for a plasma accelerator that has received a considerable amount of experimental work is shown in figure 20. This particular plasma accelerator was devised by W. H. Bostick (ref. 12). Sketch 1 of figure 20 shows two electrodes in an insulator material. A condenser is discharged through these two electrodes to produce an arc. In space the ions and electrons in the arc would come from the electrodes. The three sketches of figure 20 represent three different stages in the arc discharge. The time interval between sketches is of the order of a fraction of a microsecond.

The current in the arc induces a magnetic field as shown. The magnetic field and current interact to produce a force in a direction perpendicular both to the current and to the magnetic field. The magnetic field is stronger on the inside of the curved arc, and therefore there is a net force in the outward direction which accelerates the plasma as shown. Specific impulses up to about 20,000 seconds have been measured. Figure 21 shows a Kerr cell photograph (taken by Bostick) of a plasma about 1/2 microsecond after firing. The bright spot is at the accelerator, and the luminous area has the typical horseshoe shape shown in sketch 2 of figure 20.

A propulsion system using this plasma accelerator is shown in figure 22. Several of the plasma accelerators, each with its own condenser and switch, are connected in parallel to a high-voltage direct-current source. The system is designed so that each accelerator fires at the rate of 1000 pulses per second. The thrust from such a system would be about 1 pound per 100 plasma accelerators. The weight per kilowatt of jet power is only a small fraction of a pound for the propulsion portion of the system. The principal weight in the system would be the weight of the electric generating system. If a nuclear-electric system were used, the weight would be about $5\frac{1}{2}$ pounds per kilowatt electric power. As discussed previously, the weight penalty of the high-voltage direct-current supply is small.

The plasma accelerators are not nearly 100 percent efficient as there are unavoidable losses in heating the electrodes and in the switch. Some preliminary experiments carried out at the Lewis Research Center indicate that efficiencies of the order of 40 percent or higher could be attained. With an efficiency of 40 percent, the optimum specific impulse for the Mars journey would be about 11,000 seconds, and the weight of propellant and powerplant per pound of thrust about 7000 pounds. If an efficiency of 80 percent could be attained, these figures would become about 15,000 seconds and 5000 pounds, respectively. These values are about the same as those for the ion-accelerator propulsion system.

Photon Generators

The use of artifically generated photons is often referred to as the ultimate in jet propulsion. As yet, however, no satisfactory method of generating photons is known. If the electrical systems discussed are relied on, the specific impulse of photons is much too high. Even with 100-percent efficiency of conversion of electric power into directed photons, a nuclear-electric system of the sort considered would weigh about $3\frac{1}{2}$ million pounds to generate 1 pound of thrust. This gives a thrust-weight ratio of the order of 10^{-7}, and about 10^{-4} is needed to get to Mars in a reasonable length of time. For journeys within the solar system, therefore, it is much better on an initial weight basis to limit specific impulses to the order of 30,000 rather than 30,000,000.

For interstellar or intergalactic journeys, photon propulsion may be the ultimate solution, but a process must first be found to convert large portions of mass into directed photons. Current fission and fusion reactions contemplated for power generation convert only about 5 percent of the mass into energy.

Photon and Radioisotope Sails

Since it seems uneconomical to try to produce photons, perhaps the photons provided by the Sun could be used. A perfect reflector normal to the Sun's rays, outside the atomsphere, at the same distance from the Sun as the Earth feels a force of about 2×10^{-7} pound per square foot. This is a small pressure and accordingly requires lightweight reflectors. Figure 23 shows a plastic balloon that is silvered over the outside surface and has instruments located in the center. If the balloon were 1/2-mil-thick plastic, its weight per square foot of surface would be 3×10^{-3} pound per square foot. The ideal thrust-weight ratio, that is, the thrust-weight ratio for a section of surface normal to the Sun's rays, would be about 7×10^{-5}. The actual thrust-weight ratio would be less than half of this, since half of the balloon surface is always inoperative and some of the operating surface will not be normal to the sun.

Actually, this photon sail would not be able to escape from a satellite orbit to free space unless there were some kind of control system that would vary the orientation of the mirror as the position of the vehicle is changed. This would further add to the weight.

In view of the low thrust-weight ratio, this idea does not look too interesting for any of the manned missions being considered. The low thrust-weight ratio and the control problem make the idea uninteresting for the unmanned missions, since development of an automatic control system would require an effort that would probably be better expended elsewhere.

Another idea similar to the photon sail is the radioactive sail or alpha sail shown in figure 24. A radioisotope which is an alpha-emitter (Po^{210}, in this case) is embedded in a 0.2-mil layer of plastic which is backed by a 1-mil layer of plastic. Alpha particles are emitted in both directions, but those in one direction are stopped by the 1-mil-thick plastic backing sheet. This results in a net thrust of 10^{-6} pound per square foot, which is higher than that of the photon sail, but the weight is also higher, 9×10^{-3} pound per square foot. The net result is that the ideal thrust-weight ratio would be about 10^{-4}, about the same as that of the photon sail.

A "parachute" geometry has been shown for the radioisotope sail instead of a balloon geometry as for the photon sail. Either geometry is possible for either type of sail. The problems of control are probably greater for a parachute geometry.

Of course, some method of emitting electrons from the radioisotope sail would be required to prevent charge buildup on the vehicle. This is not shown on figure 24 for the sake of simplicity.

The alpha sail does not give any appreciable advantage in thrust-weight ratio compared with the photon sail and has at least one great disadvantage - namely, the loss in thrust with the decay of the radioisotope. Therefore, of the two schemes the photon sail looks more interesting; but, as previously observed, it is not very attractive for the missions being considered.

Recombination and Solar Ramjets

Two Earth-bound propulsion systems have been proposed over the years which, in theory, could support flight indefinitely in the Earth's rarefied upper atmosphere, without carrying chemical fuel. Discussion of the ionosphere recombination ramjet will be presented first, and then the solar-powered ramjet will be discussed briefly.

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Readers interested in more details on the recombination ramjet should also see reference 3.

Recombination ionosphere ramjet. - Above 52 miles in the Earth's atmosphere the oxygen and nitrogen of air are dissociated by the Sun's ultraviolet rays into chemically active free radicals or atoms. The idea of a recombination ramjet is to take these energetic air particles on board and to convert their energy into heat and thereby obtain thrust.

Granting for the moment that the idea is sound, the first question is: At what altitudes and flight speeds is the ionosphere chemical energy useful for propulsion? A preliminary analysis showed that even an all-supersonic ramjet would require more energy for providing lift and overcoming drag than is available at any ionosphere altitude. Therefore, a recombination ramjet is considered traveling at orbital velocity where only drag need be overcome.

The thrust that could theoretically be generated from the recombination energy available is compared with the drag for several nacelle configurations in figure 25. The thrust parameter is the thrust divided by the ramjet inlet area and the ambient air density. Similarly, the external drag is divided by the inlet area and density for direct comparison. The crosshatched area indicates the probable limits on the energy available at altitudes from 300,000 to 700,000 feet. The range of energies shown is the result of uncertainty in ionosphere physical properties. The external drags are shown for a ramjet length of 100 feet. The far right curve is the drag for a truncated cone with a positive angle of 2° and an inlet radius of 10 feet. Similarly, the next curve is for a more promising configuration: -4.3° angle and 10-foot radius. Finally, the far left curve is for a nacelle with a -8.6° cone half-angle and 20-foot inlet radius.

Only the -8.6° nacelle gives a drag appreciably lower than the probable maximum thrust. Therefore, an engine for this nacelle will be considered in more detail with a thermodynamic cycle.

Figure 26 summarizes the results of a cycle analysis. The nacelle geometry is shown to scale. A low ionosphere altitude of 328,000 feet was chosen for this example. At this altitude only oxygen is dissociated into free radicals. The cycle involves swallowing this energetic air and exhausting a hot recombined air jet. The cycle is illustrated in figure 27, which shows stations 1, 2, and 3 on a static temperature-pressure plot. Station 1 is the shock-free inlet station; station 2 is the internal throat station; station 3 is the nozzle-exit position. A frozen-composition compression from about 5×10^{-7} to 6×10^{-3} atmosphere brings the inlet air to a temperature-pressure condition at station 2 where it theoretically can be converted adiabatically and isothermally to chemical equilibrium. The exhaust expansion from station 2 to 3 is assumed to follow chemical equilibrium. Notice that from station 2 down to about 10^{-4} atmosphere, recombination is proceeding, causing the unusual temperature-pressure relation in this region. Station 3 is at a considerably higher pressure than ambient, because the nacelle geometry for this example does not allow full expansion. (If expansion to ambient pressure station 4 were possible, the engine efficiency would be 85 percent.) The resulting overall engine efficiency, shown in figure 26, is 22 percent; the thrust is an order of magnitude greater than external drag for this example.

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Actual hardware design of this engine involves at least two serious problems. The inlet requires a very large contraction ratio in a short length; but perhaps a multiple diffuser could do this job. Also, the chemical kinetics of recombination are not understood well enough today for proper internal flow design. However, if interest in a large, low-flying satellite is great enough, none of these problems appear unsolvable.

Solar-powered ramjet. - Another device attractive in principle is the solar-powered ramjet. It was mentioned earlier that the Sun supplies the Earth with about 100 watts per square foot of normal surface as radiant energy. Perhaps this solar energy can be used directly for high-altitude satellite propulsion in a ramjet.

Naturally, the problem is how to get this radiant energy into the airstream for the heat cycle. A ramjet using air as a working fluid cannot absorb any appreciable fraction of this solar energy directly. Therefore, a convective heat exchanger must be used in conjunction with a solar-energy collector or lens. That is, the collector would heat a metal heat-transfer surface, and the air passing over this surface would be heated to supply thrust.

The basic problem of such a device is heat transfer. In a convective heat exchanger, the ΔT for heat transfer is the temperature of the wall minus the adiabatic wall temperature. The adiabatic wall temperature exceeds the material limit on today's metals at flight Mach numbers greater than about 7. Therefore, the solar-powered ramjet is not useful for satellite-sustaining.

COMPARISON OF ELECTRIC PROPULSION SYSTEMS WITH CHEMICAL AND NUCLEAR ROCKETS

The more promising propulsion systems discussed will now be compared with chemical and nuclear rockets for several typical missions. Figure 28 shows the powerplant plus propellant weight as a function of required sustaining time for a satellite-sustainer application. The requirement is the production of a sustaining thrust of 0.05 pound. This thrust is adequate to overcome the drag of a 6-foot-diameter hemisphere-cylinder satellite about 30 feet long at an altitude of about 100 miles. The thrust could also be used for orbit control. The two curves for rockets, with specific impulses of 300 and 1000 seconds, include only the propellant required and not the powerplant. It is doubtful whether rockets could be designed to achieve these specific impulses at the very low thrust level indicated. Such rockets would probably be operated at higher thrust levels for short periods of time, but the overall propellant consumption would be comparable to that shown. The electric systems require an electric output of about 10 kilowatts at a specific impulse of about 10,000 seconds to generate the required 0.05 pound of thrust. The weights shown are minimum values, without penalty in shielding or for part-time-in-Sun operation. Depending on the magnitude of these penalties, the crossover points relative to the hypothetical rockets will shift toward larger sustaining times. No precise value can therefore be given for the sustaining time at which the electric systems become superior as to weight, but it is clear from the slopes of the curves that the electric systems will eventually become superior. In particular, if a satellite is to be maintained aloft indefinitely, the resupply weights are much less for the electric than for the chemical system, owing to the much higher specific impulse.

Figure 29 compares the initial weight required for an unmanned one-way trip to Mars from a satellite orbit near Earth to a satellite orbit around Mars. An instrument payload of 2000 pounds is allowed. Two chemical rockets and two nuclear-electric ion systems are considered. The nuclear heat-transfer rocket has been omitted from comparison for this mission because no estimates have been made of such low weight nuclear motors. The comparison shows that the advanced chemical rocket ($I = 420$ sec) is capable of undertaking this mission with little weight penalty relative to the nuclear-electric system. However, the 2000-pound payload for the chemical rocket must include an electric powerplant for transmitting the data, whereas the powerplant used for propulsion can serve this function for the electric rocket system. The effective payload for the electrically propelled vehicle is therefore greater

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by the proportion of its powerplant that is needed for the communication system.

Figure 30 compares initial weight required for a full-scale manned trip to the Moon with landing and exploration equipment. The basic payload, which includes all items carried throughout the trip, is taken to be 10,000 pounds. An additional subsistence allowance of 10 pounds per man-day was considered, and a landing and exploration equipment weight of 16,000 pounds was assumed. The initial weight comparison is for the two chemical rockets of figure 29, the nuclear-electric-ion systems, and a low-pressure high-specific-impulse nuclear rocket. The unshielded nuclear-electric system is approximately the same weight as the solar turboelectric, so that this column serves a dual purpose. A specific powerplant weight of 10 pounds per kilowatt was assumed for the nuclear-electric system. This weight would be attainable if the conversion of electric power to jet power is accomplished with 70-percent efficiency and if the thrust generator weighs 2 pounds per kilowatt. These values, as previously indicated, appear to be attainable. The comparison shows that the largest drop in required initial weight occurs in going from the $I = 300$ chemical rocket to the $I = 420$ chemical rocket. There is little further gain in going to the nuclear rocket because of the high motor weight and, consequently, the reduced staging advantage. Some additional weight reduction is possible by going to the nuclear-electric-ion system, but this system is unattractive for the manned Moon mission because of the long travel time required. The advanced chemical rocket is therefore capable of undertaking this mission without excessive weight penalty.

The same systems are compared in figure 31 for a similar manned expedition to Mars. For this mission, a basic payload of 50,000 pounds and additional landing and exploration equipment of 60,000 pounds are assumed. A substantial weight reduction, even over that obtained with the advanced chemical rocket, is possible for this mission by going to the nuclear rocket or the nuclear-electric-ion system. A point worth noting in this comparison is that the total initial payload, consisting of the basic payload, the landing and exploration equipment, and the subsistence supplies, is about 200,000 pounds. The initial gross weight for the nuclear rocket and nuclear-electric systems is therefore only $2\frac{1}{2}$ to 3 times the initial payload weight. This means that there is not too much margin left for reducing the gross weight of an expedition of this magnitude, and the nuclear propulsion systems considered are, in fact, very good systems for this mission.

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CONCLUSIONS

Auxiliary Electric Power

Systems using solar energy (solar batteries and solar turboelectric systems) involve the least weight for power requirements up to a few kilowatts, provided almost-continuous operation in the Sun is possible. If only half time is spent in the Sun, a number of systems are competitive, including radioisotope hydrogen-oxygen cells and radioisotope turboelectric systems (for durations comparable to the half-life of the isotope). The solar turboelectric systems can be used only at altitudes above about 300 miles since the drag of the required balloon collector is excessive below this altitude. The nuclear turboelectric system without shielding is competitive in this range of power, but shielding requirements, particularly for manned vehicles, may rule it out. Chemical batteries are competitive weightwise only for durations of operations of the order of a few days. The required voltage must also be considered in selecting auxiliary power systems.

Satellite Sustainers and Orbit Control

For periods of operation of the order of 100 to 200 days or less, a chemical rocket can provide the required propulsive energy without excessive weight penalty relative to electric systems. Particularly, if rapid orbit changes are required, the chemical rocket seems to be the only feasible propulsion system. For very long durations, or for permanent satellites, electric propulsion systems using solar energy or nuclear energy require less initial weight or resupply weight than chemical rockets. The solar turboelectric system is restricted to altitudes above about 300 miles, and the solar batteries are limited in the voltage attainable with a practical arrangement. Consequently, the nuclear turboelectric system with ion or plasma accelerators seems most satisfactory for this application if shielding weight can be kept low.

The recombination ramjet may be feasible for sustaining satellites indefinitely at altitudes near 60 miles if the powerplant is made sufficiently large. However, many serious questions remain concerning the possibility of designing the required short inlet with very large contraction and concerning the magnitudes of the recombination rates.

Lunar and Mars Journeys

Many missions involving trips to the Moon and Mars can be accomplished without excessive weight penalty with high-performance chemical rockets ($I \approx 420$ sec). These missions include one-way instrumented journeys to the Moon and Mars and manned trips to the Moon. Electric propulsion systems seem undesirable for the Moon trip because of the long times required for the journey compared with those required for high-thrust rockets. For manned trips to Mars, however, electric propulsion systems require only moderately more time for the complete journey than the impulse rocket, and their advantage in initial weight becomes greater and greater as the size of the expedition increases. Of the electric systems considered, the nuclear turboelectric, the solar turboelectric, and possibly the fusion-powered systems are capable of supplying the required electric power with sufficiently low weight. Of the thrust generators considered, the ion-electric accelerator appears to be most promising on the basis of current technology.

The low-pressure, high-specific-impulse nuclear rocket is competitive with electric systems for large-scale Mars expeditions and has the advantage of higher thrust-weight ratio. It has the disadvantages that much higher temperatures are required than in the electric systems and that hydrogen must be used to attain the required high specific impulse. The latter requirement imposes severe storage difficulties for long-duration journeys.

Another advantage of the electric system over both the nuclear and chemical rockets is the resupply advantage. Since the electric system has the higher specific impulse, its propellant replacement weight is much less than that of the chemical and nuclear rockets.

REFERENCES

1. Moeckel, W. E.: Propulsion Methods in Astronautics. Int. Series on Aero. Sci and Space Flight. Vol. 2. Pergamon Press, 1959, pp. 1078-1097.

2. English, Robert E., Slone, Henry O., Bernatowicz, Daniel T., Davison, Elmer H., and Lieblein, Seymour: A 20,000-Kilowatt Nuclear Turboelectric Power Supply for Manned Space Vehicles. NASA MEMO 2-20-59E, 1959.

3. Baldwin, Lionel V., and Blackshear, Perry L.: Preliminary Survey of Propulsion Using Chemical Energy Stored in the Upper Atmosphere. NACA TN 4267, 1958.

4. Veselovski, V. I.: Radiation-Chemical Processes in Inorganic Systems. Vol. 7 of Peaceful Uses of Atomic Energy, Aug. 1955, pp. 599-609.

5. Grimminger, G.: Probability That a Meteorite Will Hit or Penetrate a Body Situated in the Vicinity of the Earth. Jour. Appl. Phys., vol. 19, no. 10, Oct. 1948, pp. 947-956.

6. Whipple, Fred L.: Meteoric Phenomena and Meteorites. Ch. X of Physics and Medicine of the Upper Atmosphere, Univ. New Mexico Press (Albuquerque), 1952, pp. 137-170.

7. Ehricke, Krafft A.: The Solar Powered Space Ship. Preprint No. 310-56, Am. Rocket Soc., 1956.

8. Becker, J. A.: The Life History of Adsorbed Atoms, Ions and Molecules. Annals N. Y. Acad. Sci., vol. 58, 1954, pp. 723-740. (See also Trans. Am. Electrochem. Soc., vol. 55, 1929, pp. 153-173.)

9. Langmuir, Irving, and Kingdon, K. H.: Thermionic Effects Caused by Vapors of Alkali Metals. Proc. Roy. Soc. (London), ser. A, vol. 107, 1925, pp. 61-79. (See also Science, vol. 57, 1923, pp. 58-60.)

10. Datz, Sheldon, and Taylor, Ellison H.: Ionization on Platinum and Tungsten Surfaces. I - The Alkali Metals. Jour. Chem. Phys., vol. 25, no. 3, Sept. 1956, pp. 389-394.

11. Stuhlinger, E.: Possibilities of Electrical Space Ship Propulsion. Extract from Proc. Int. Astronautical Cong., 1954, pp. 100-119.

12. Bostick, Winston H.: Experimental Study of Ionized Matter Projected Across a Magnetic Field. Phys. Rev., vol. 104, no. 2, Oct. 15, 1956, pp. 292-299.

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TABLE I. - ESTIMATED WEIGHTS FOR ION ENGINES SUITABLE FOR MARS MISSION

ION SOURCE	EFFICIENCY, %	OPTIMUM IMPULSE, SEC	PROPELLANT + POWERPLANT WEIGHT PER UNIT THRUST
4 LB/KW AT 15,000-SEC IMPULSE	80	23,000	5,700
	40	18,000	8,500
WEIGHTLESS	80	18,000	5,000
LINEAR WITH IMPULSE; 3 LB/KW	80	14,000	6,400

CS-14637

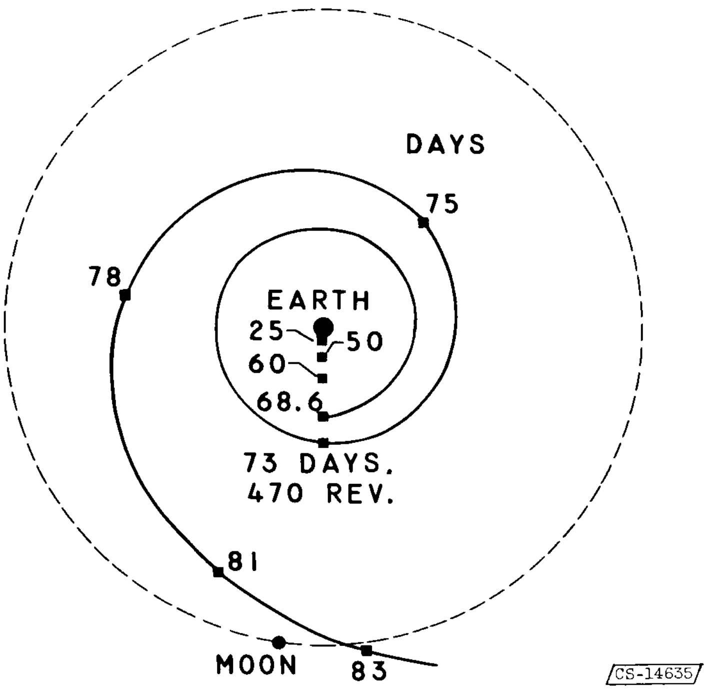
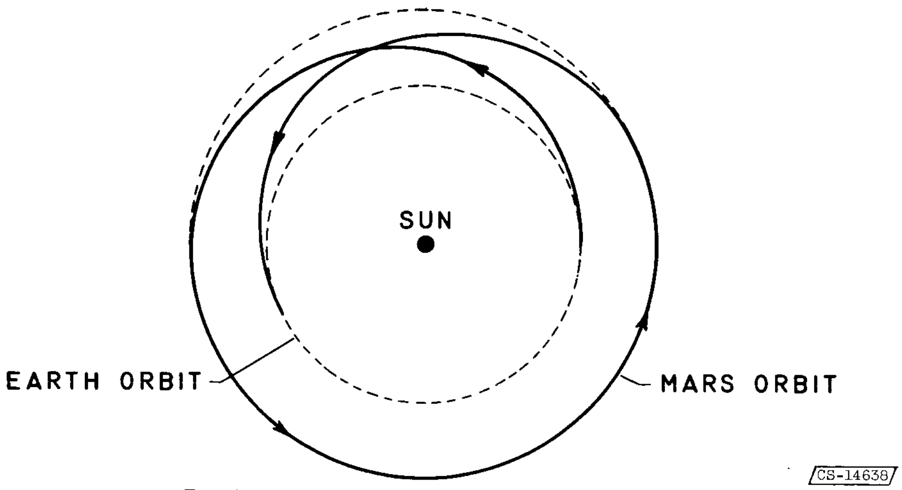


Figure 1. - Constant-thrust trajectory from satellite orbit. Thrust-weight ratio, 10^{-4}.

E-854



TIME IN DAYS FOR MARS JOURNEY

SYSTEM	ESCAPE FROM EARTH ORBIT	COAST	DESCENT TO MARS ORBIT	WAIT	TOTAL
LOW THRUST ($F/W_0 = 10^{-4}$)	127	268	85	245	1205
IMPULSE ROCKET	—	268	—	415	951

Figure 2. - Flight path to Mars.

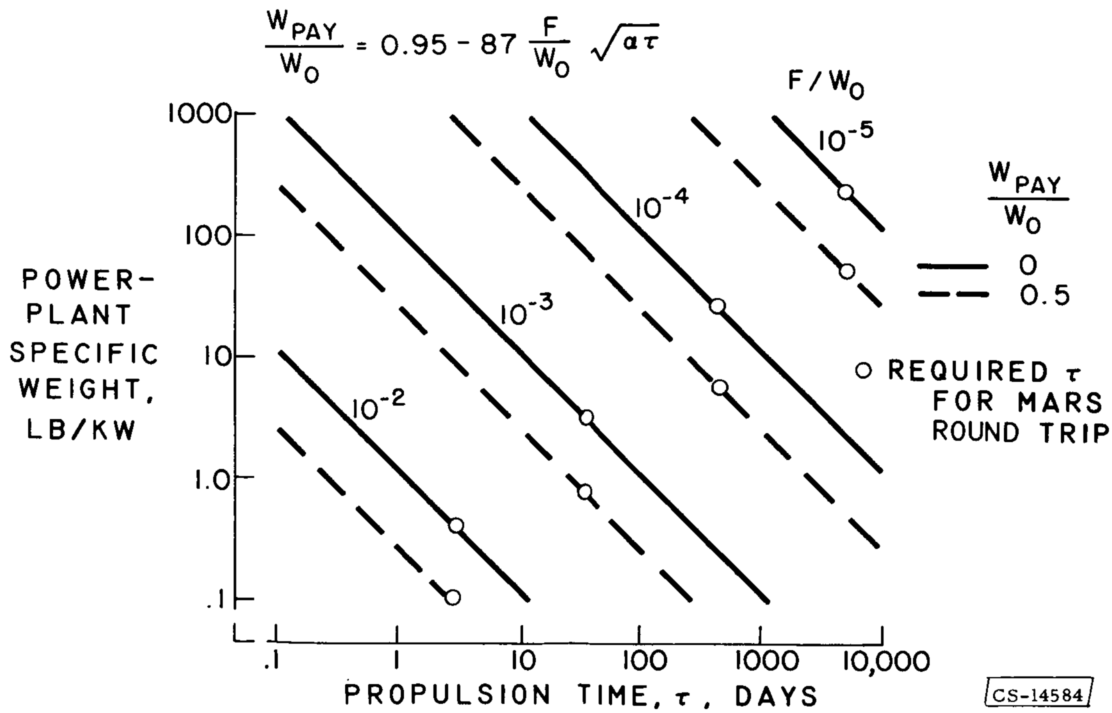


Figure 3. - Effect of powerplant weight on payload and propulsion time for optimum specific impulse.

E-854

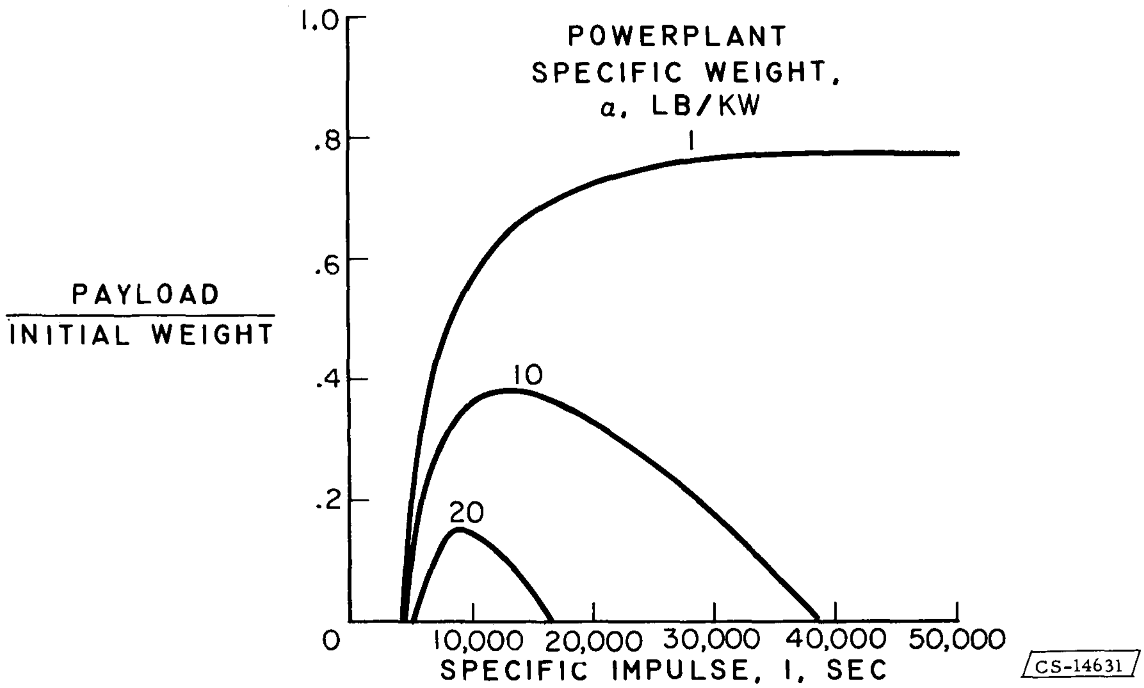


Figure 4. - Effect of nonoptimum specific impulse on payload for Mars trip. Thrust-weight ratio, 10^{-4}.

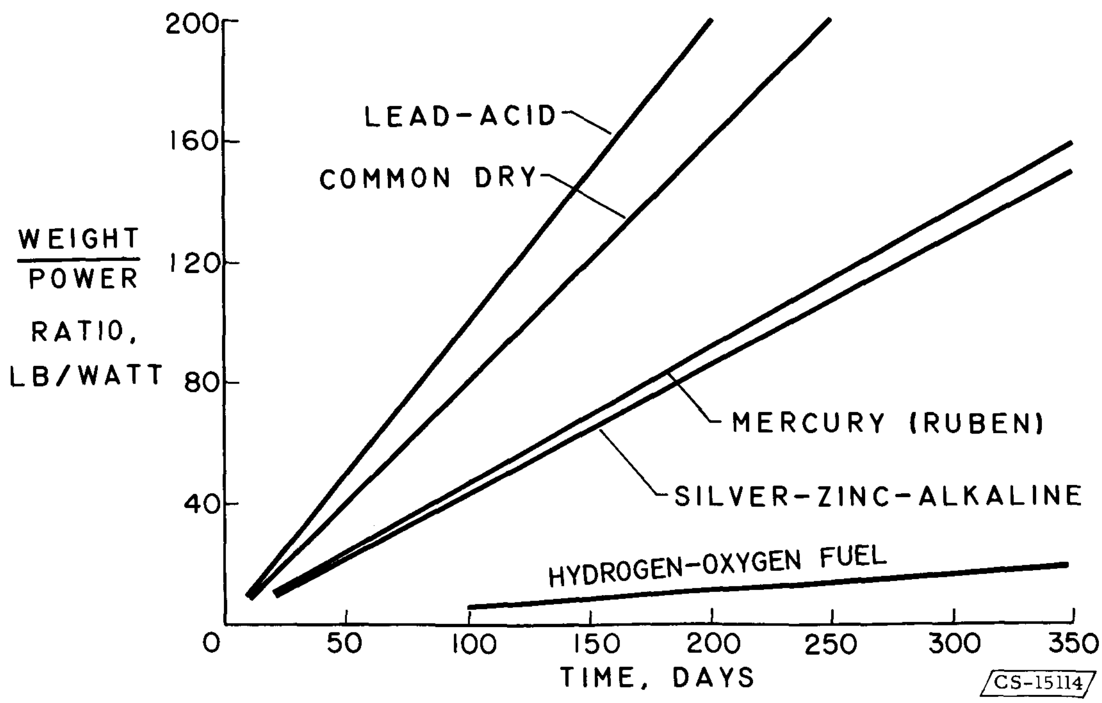


Figure 5. - Comparison of weight-power ratios (for commercial chemical batteries at various load times).

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CL-5

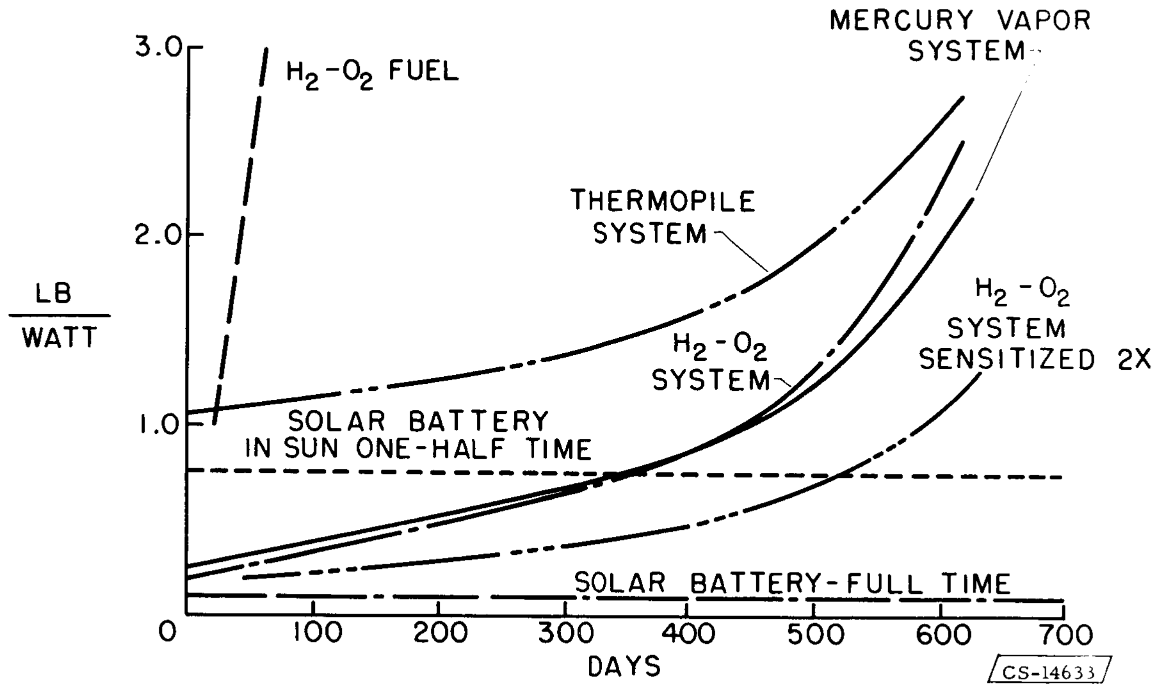


Figure 6. - Low-power electric source (continuous-operation Po^{210} energy source).

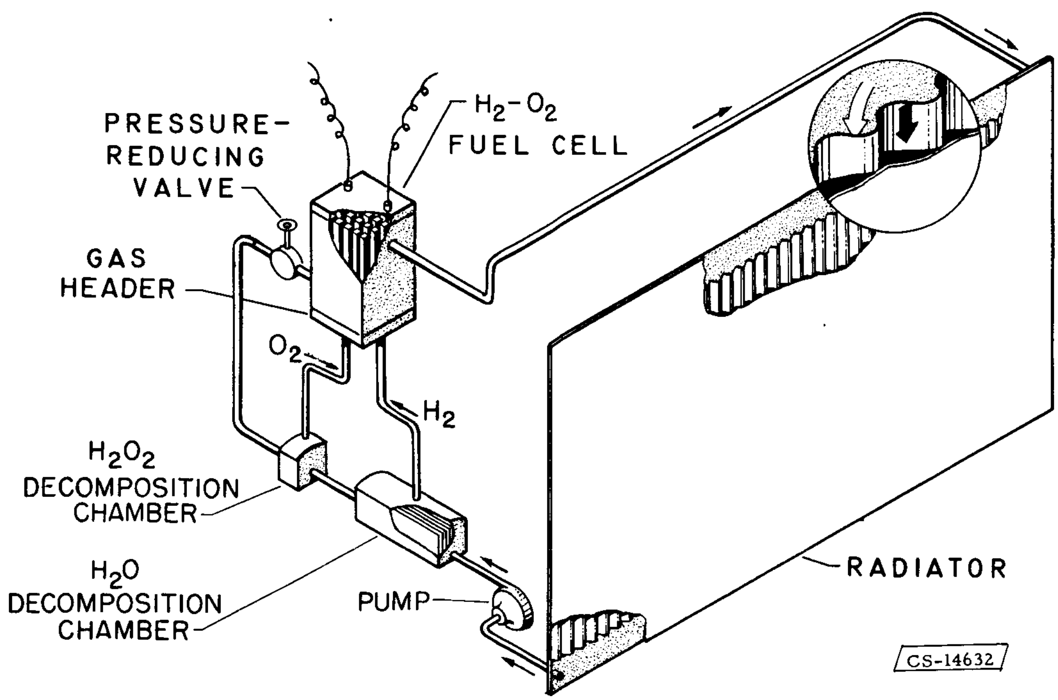
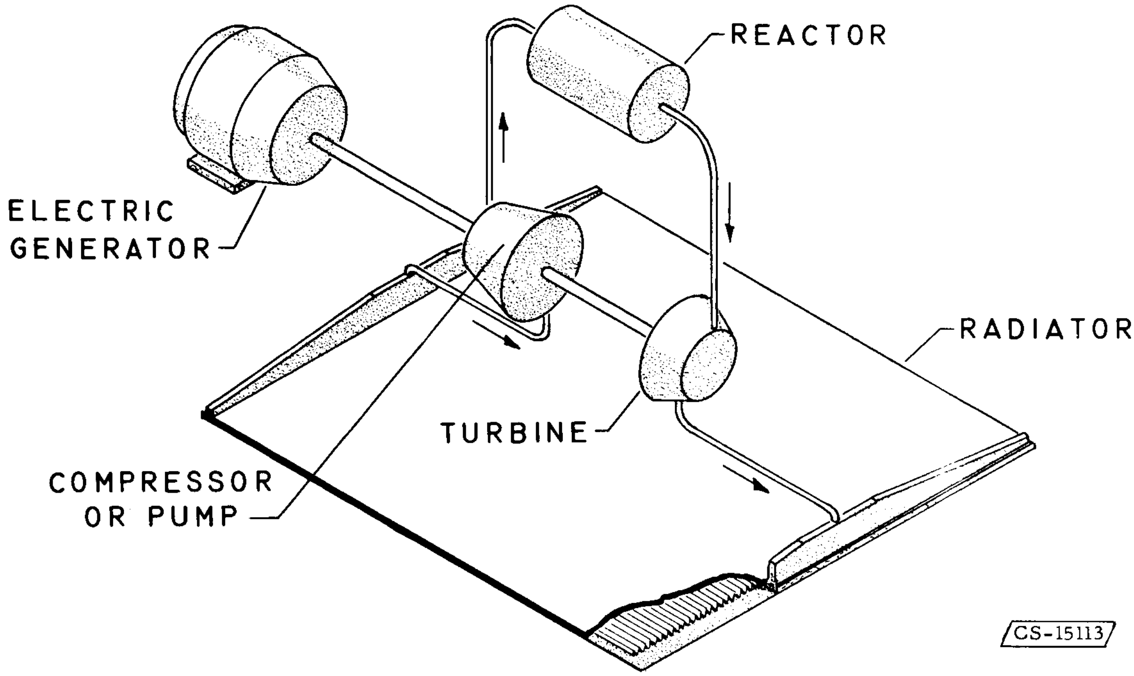
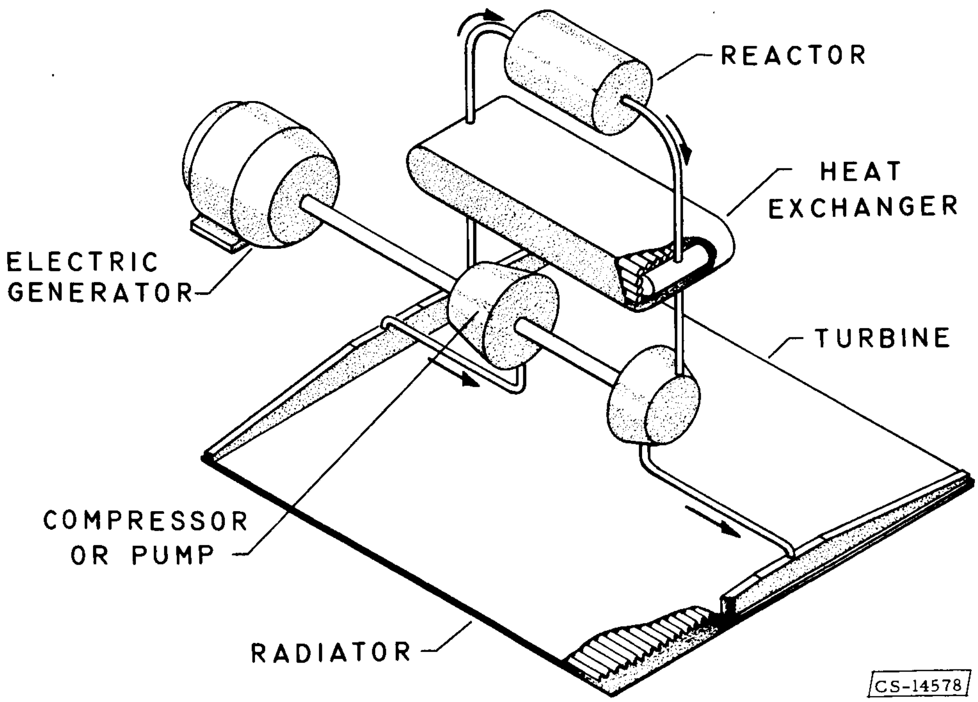


Figure 7. - Hydrogen-oxygen radioisotope power supply.



(a) Single loop.



(b) Two loop.

Figure 8. - Simplified cycle arrangement.

E-854

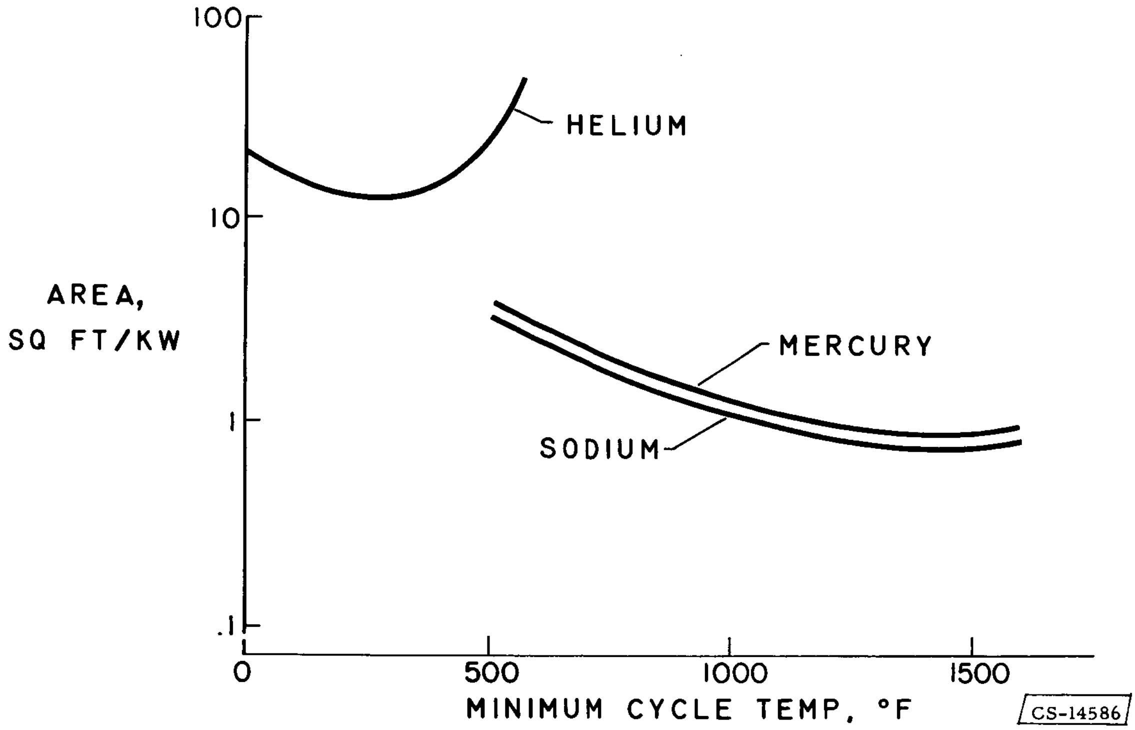


Figure 9. - Radiator area. Turbine-inlet temperature, 2040 °F; compressor and turbine efficiencies, 0.8.

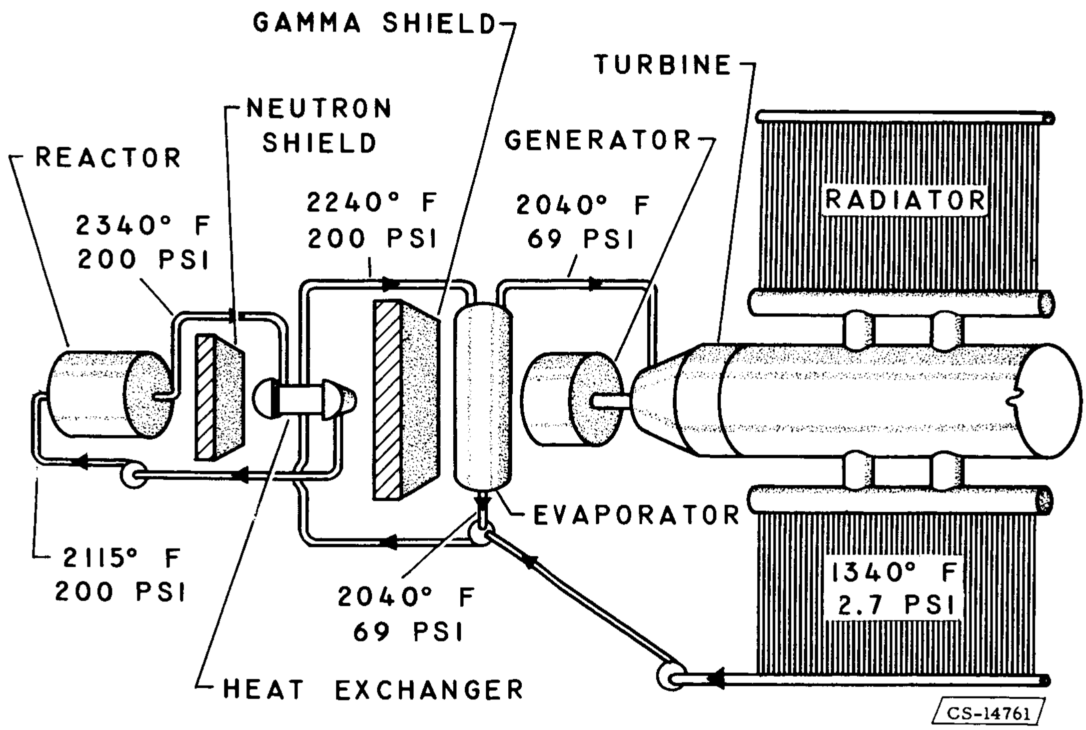


Figure 10. - Nuclear turboelectric power system using sodium vapor.

CL-5 back

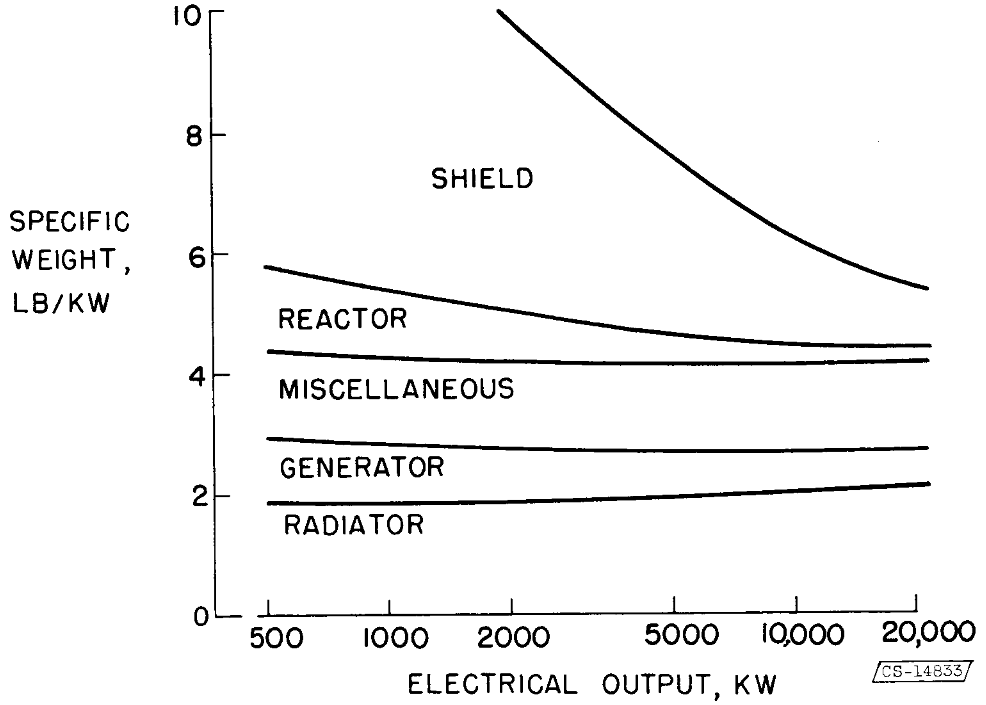


Figure 11. - Weight of power supply.

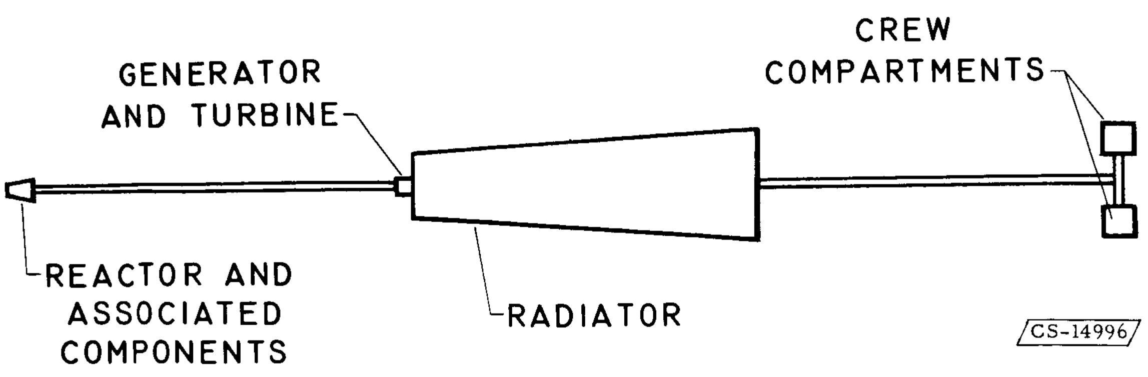


Figure 12. - Hypothetical space vehicle.

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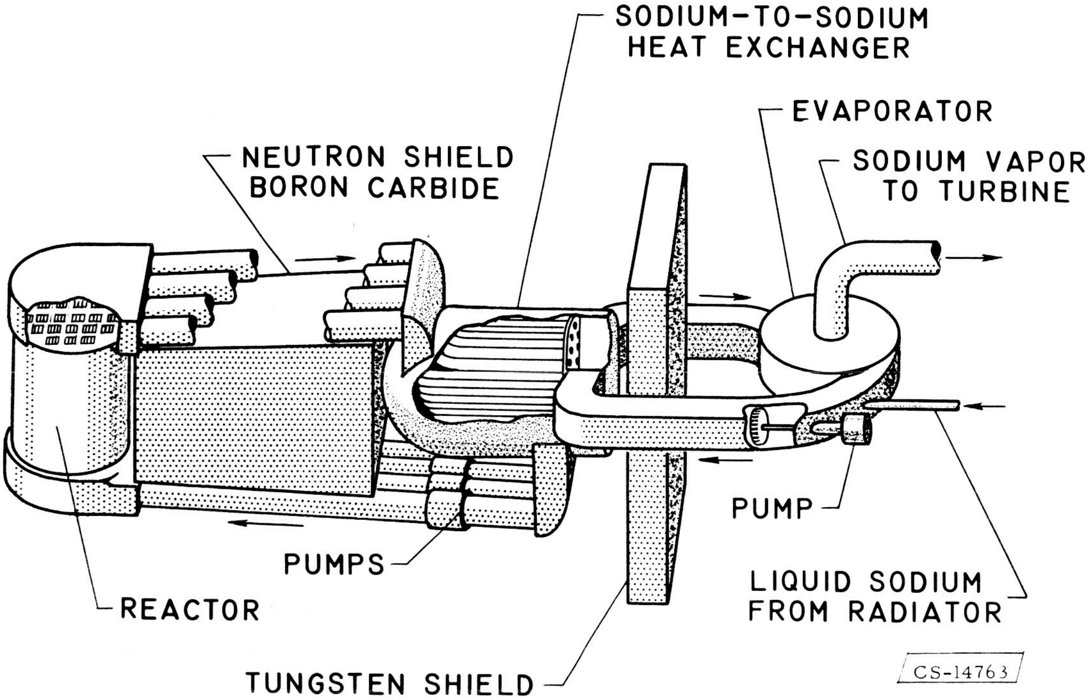
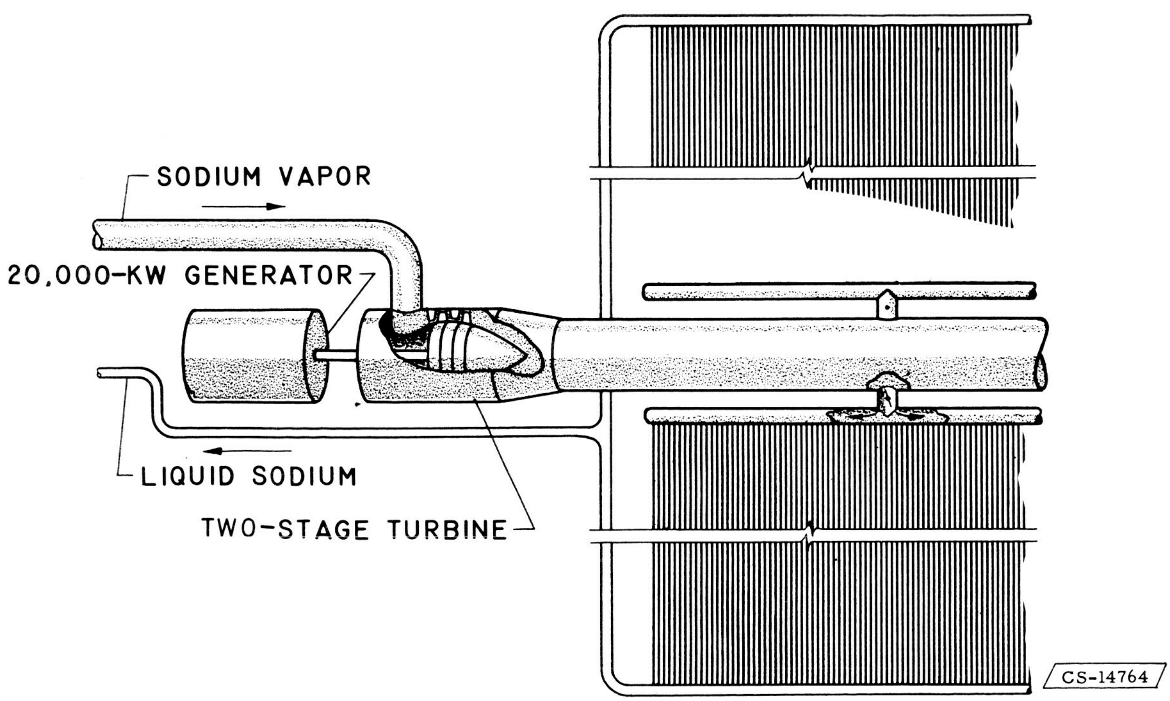
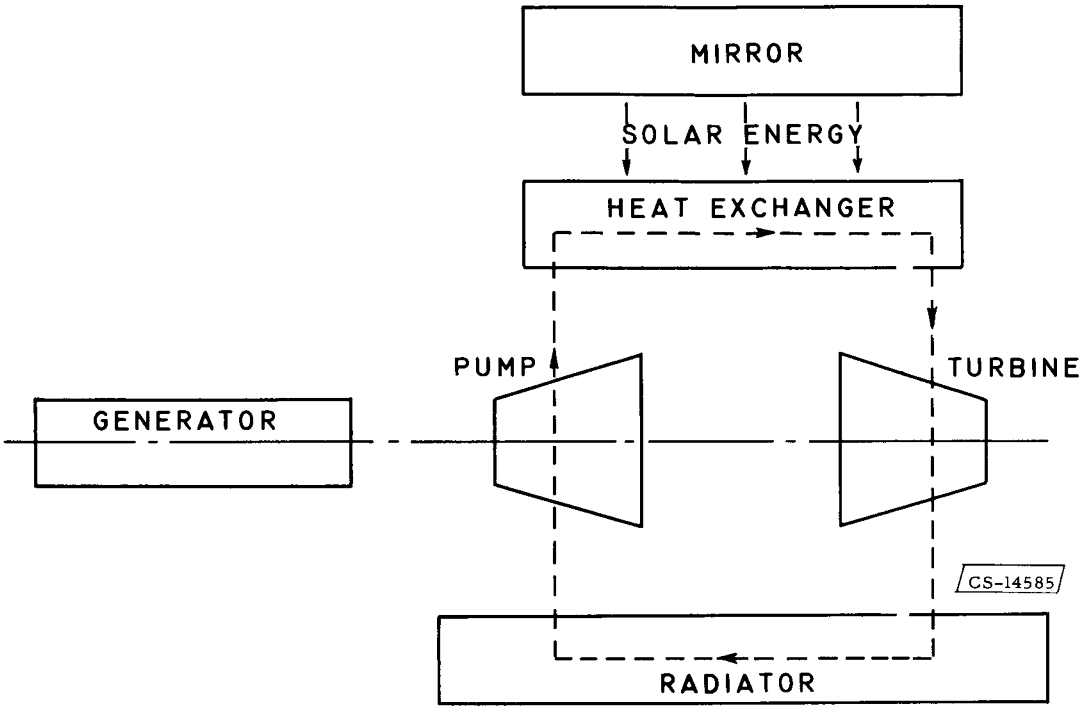


(a) Reactor.

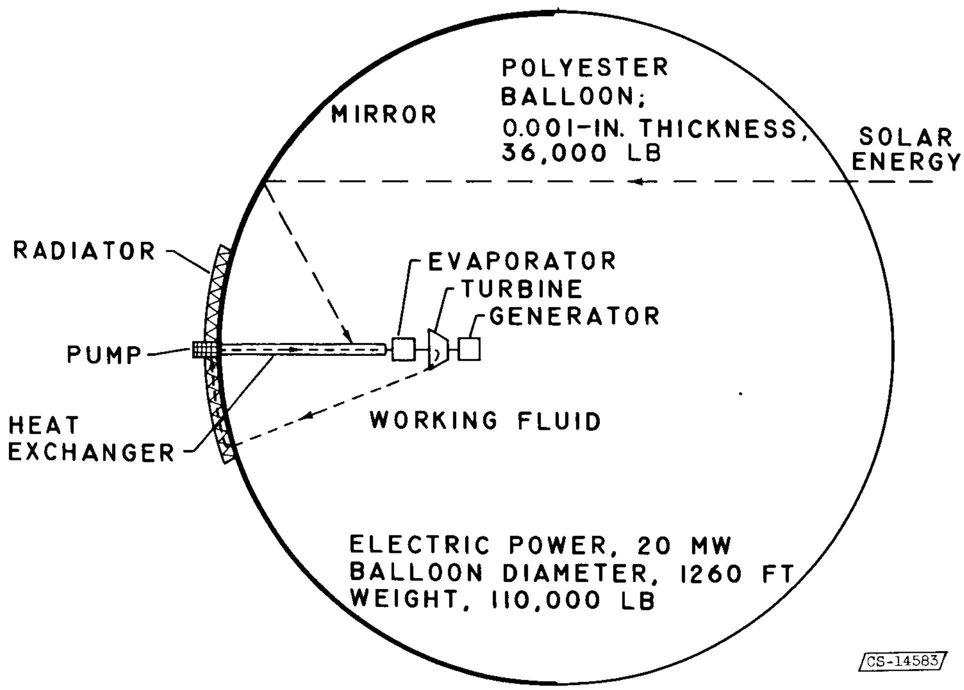


(b) Turbine.

Figure 13. - Space vehicle powerplant.



(a) Block diagram.



(b) Mirror arrangement.

Figure 14. - Solar turboelectric power supply.

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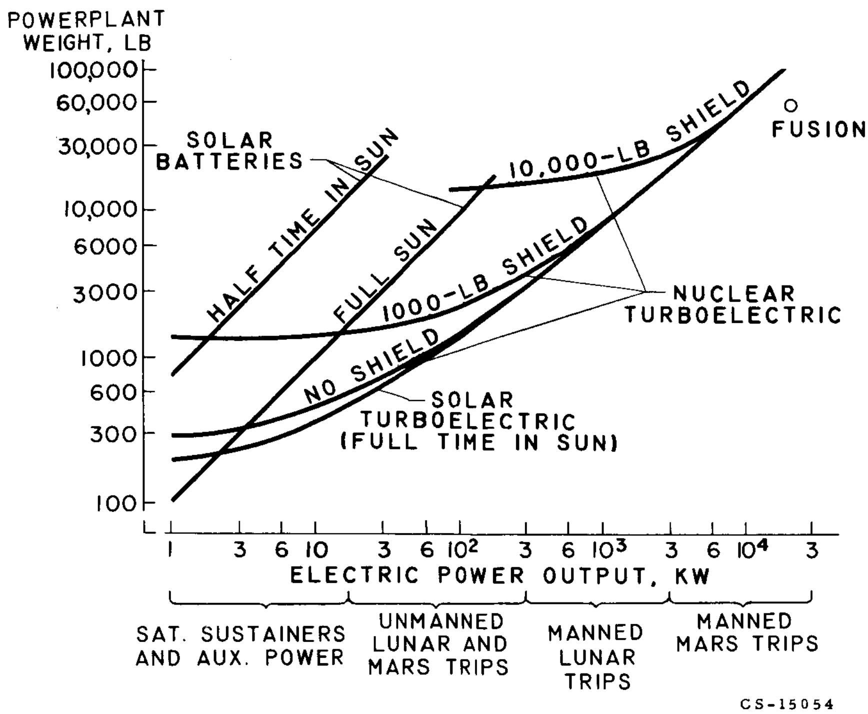

Figure 15. - Estimated weight of electric-power generators.

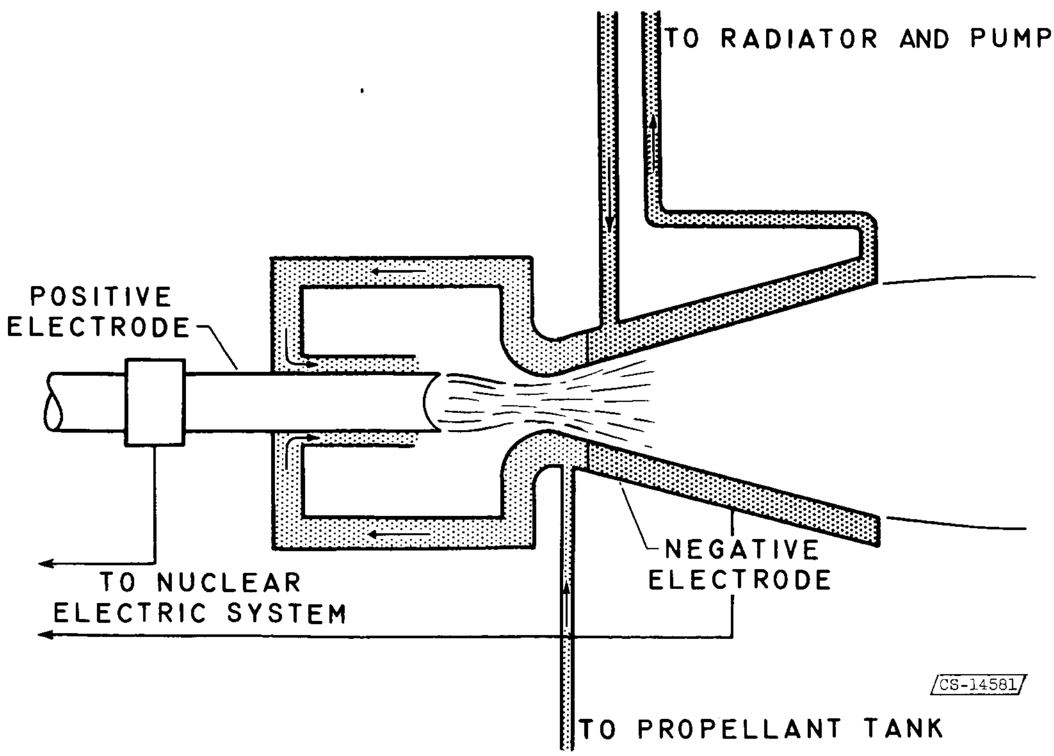

Figure 16. - Arc-jet propulsion system.

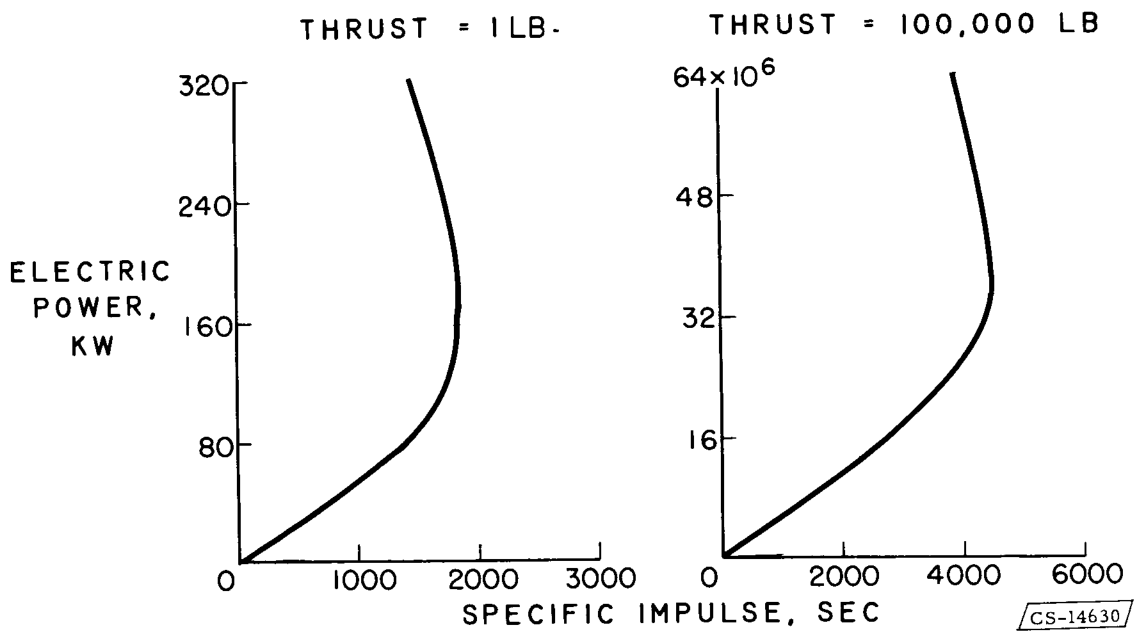

Figure 17. - Arc chamber performance. Chamber pressure, 10 atmospheres; propellant, hydrogen.

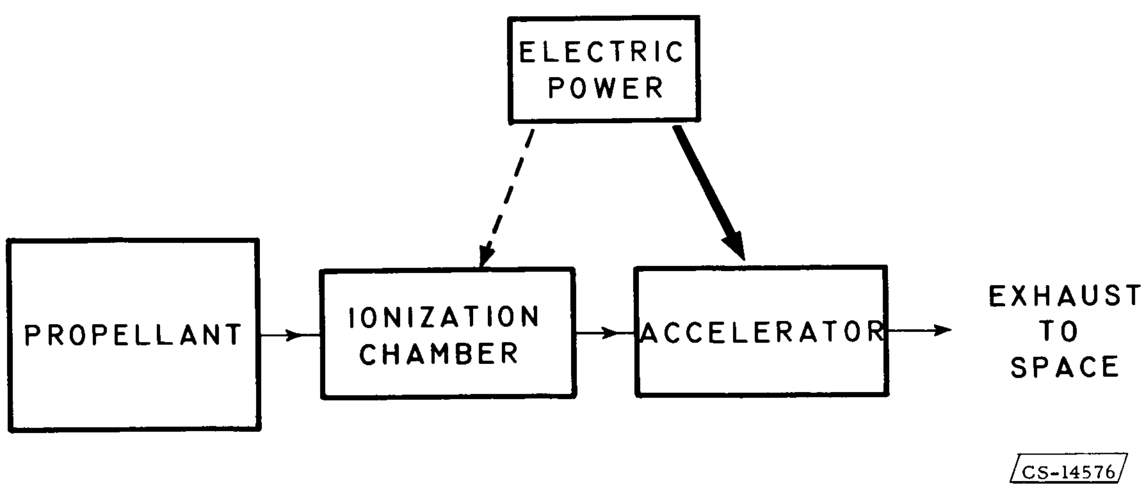

Figure 18. - Components of ion thrust system.

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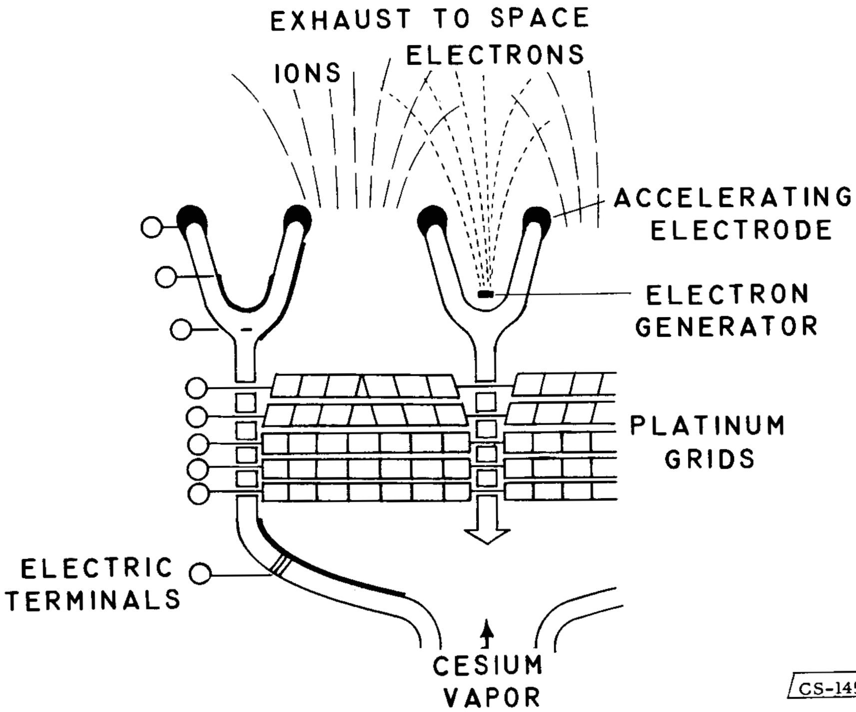


Figure 19. - Ion and electron source (Stuhlinger, ref. 11).

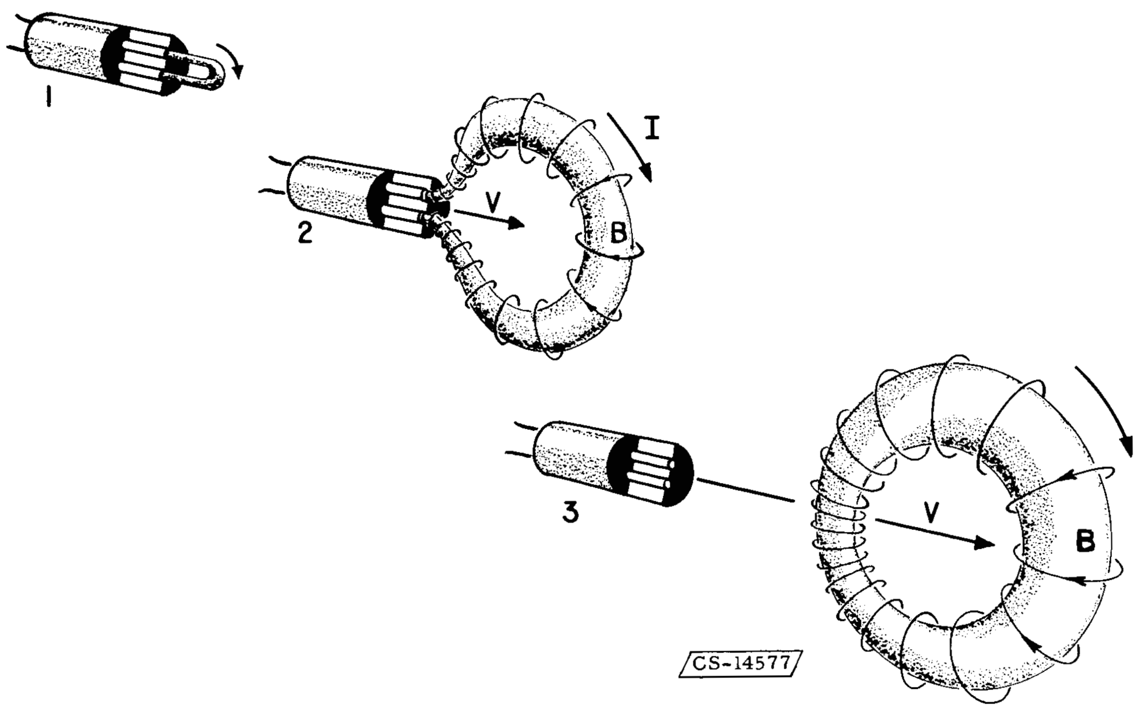


Figure 20. - Bostick's plasma accelerator (ref. 12).

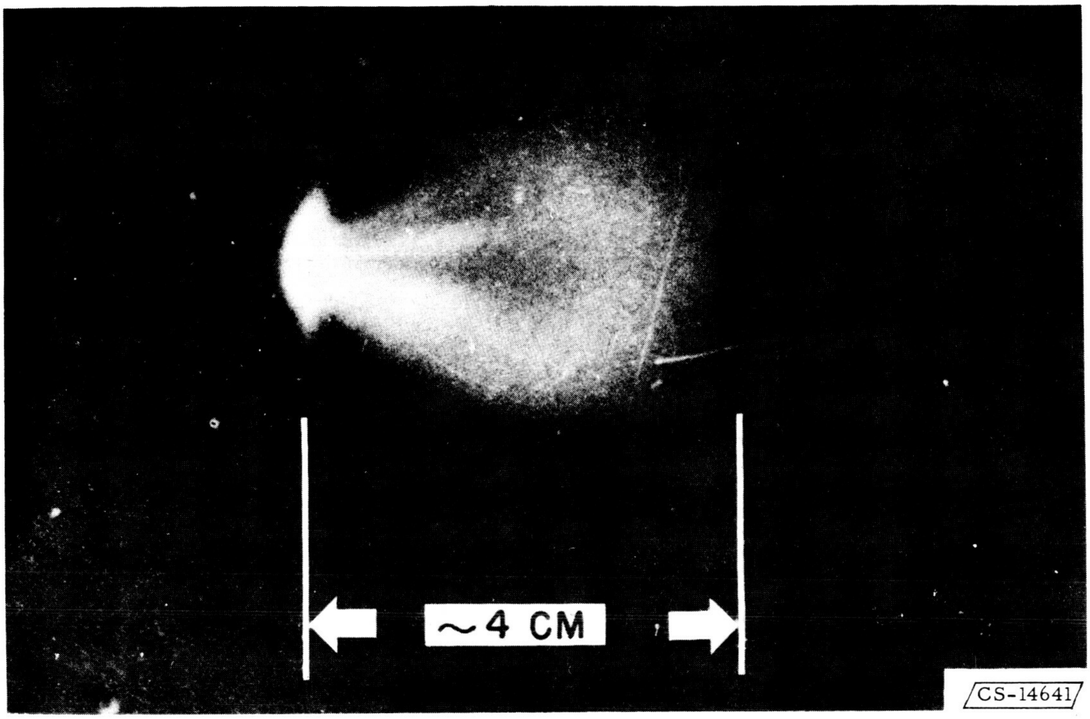
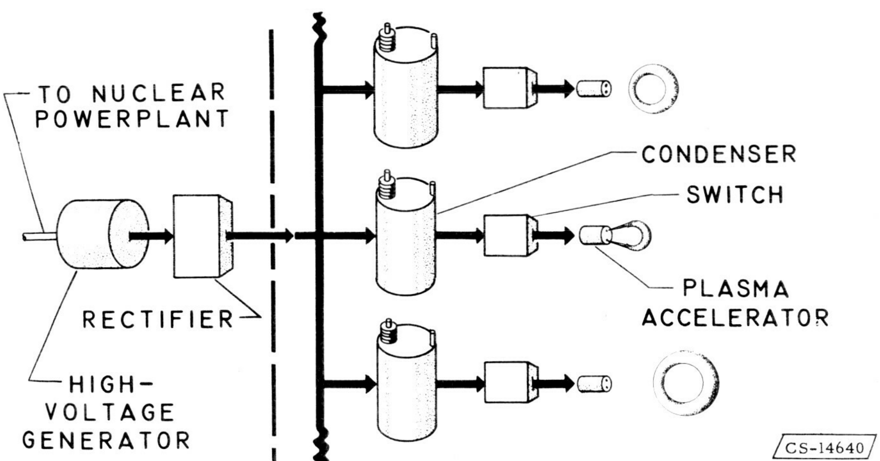


Figure 21. - Plasma 0.5 microsecond after firing.



ACCELERATOR EFFICIENCY, %	MARS JOURNEY: OPTIMUM SPECIFIC IMPULSE	MARS JOURNEY: WEIGHT OF POWERPLANT AND PROPELLANT PER LB OF THRUST
0.40	11,000	7000
0.80	15,000	5000

Figure 22. - Plasma-accelerator propulsion system.

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CL-6 back

THRUST/SQ FT	2×10^{-7}
WEIGHT/SQ FT (t = 0.0005")	3×10^{-3}
THRUST/WEIGHT (IDEAL)	7×10^{-5}

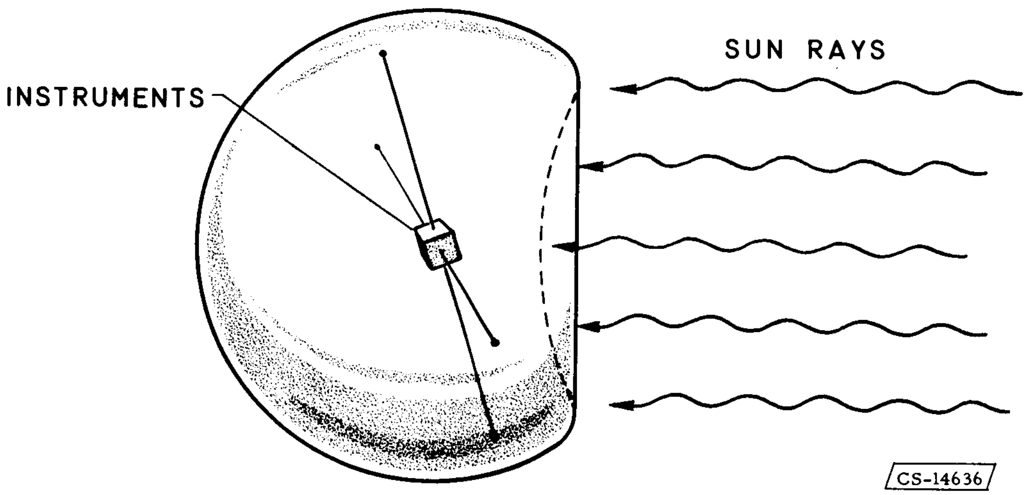

Figure 23. - Photon sail.

THRUST/SQ FT	1×10^{-6}
WEIGHT/SQ FT (t = 0.0012")	9×10^{-3}
THRUST/WEIGHT (IDEAL)	1×10^{-4}

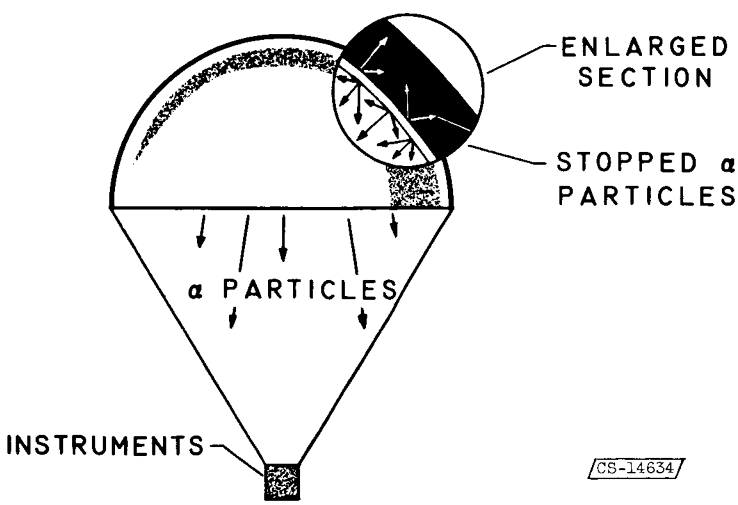

Figure 24. - Radioisotope sail.

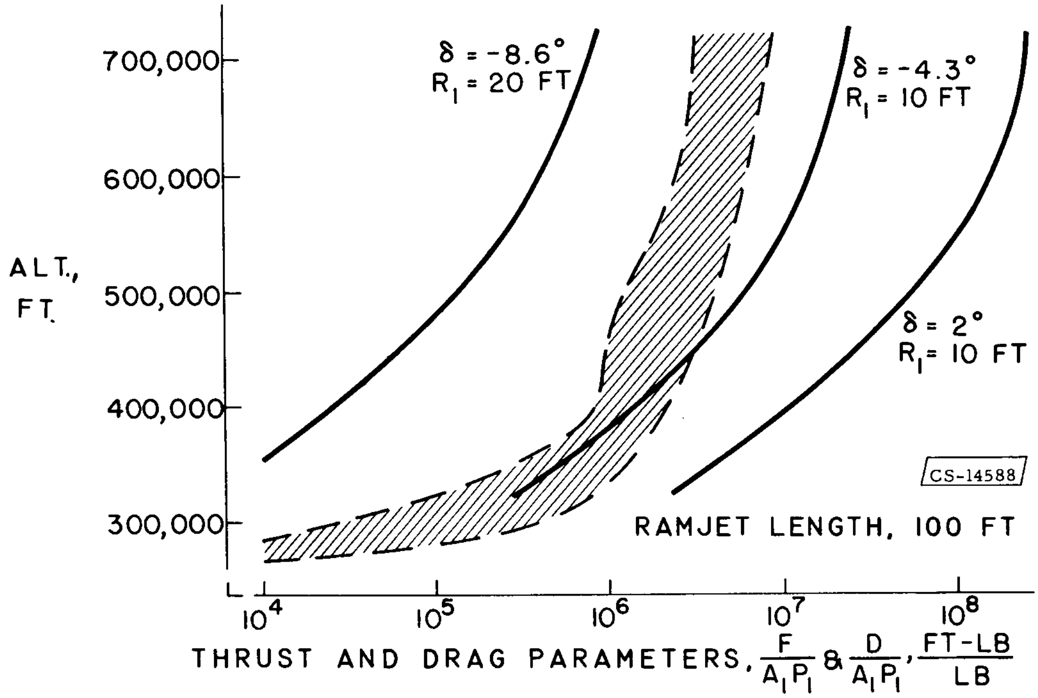

Figure 25. - Comparison of energy available with energy required in ionosphere orbits at various altitudes.

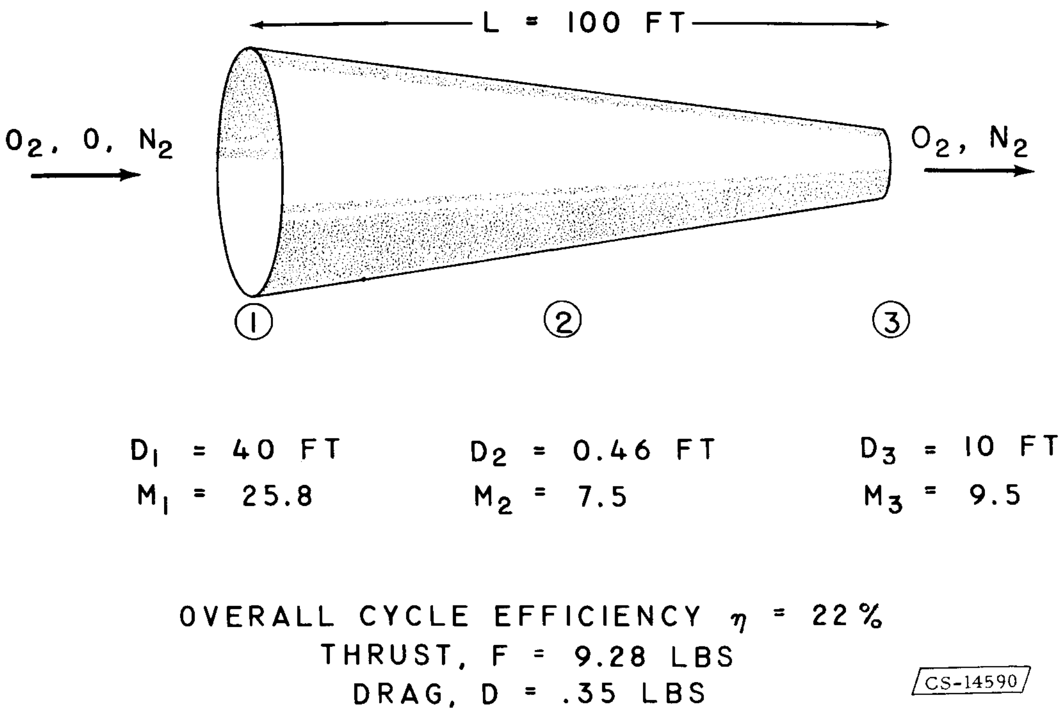

Figure 26. - One possible ionosphere ramjet orbiting at 328,000 feet. Overall cycle efficiency, 22 percent; thrust, 9.28 pounds; drag, 0.35 pound.

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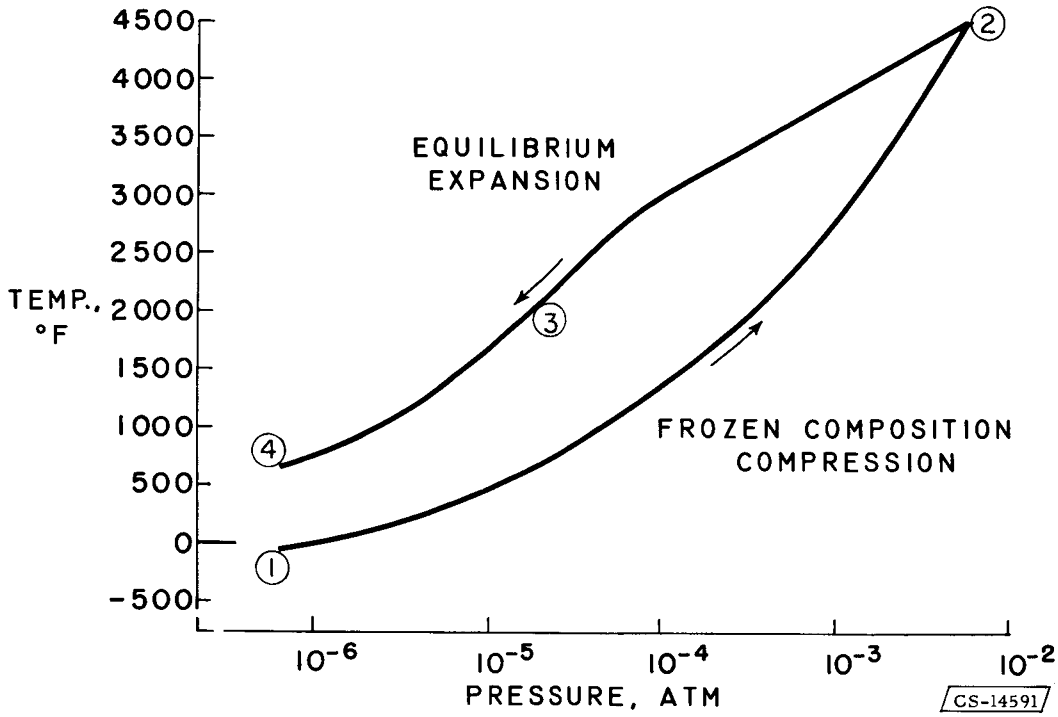

Figure 27. - Thermodynamic cycle for ionosphere ramjet orbiting at 328,000 feet.

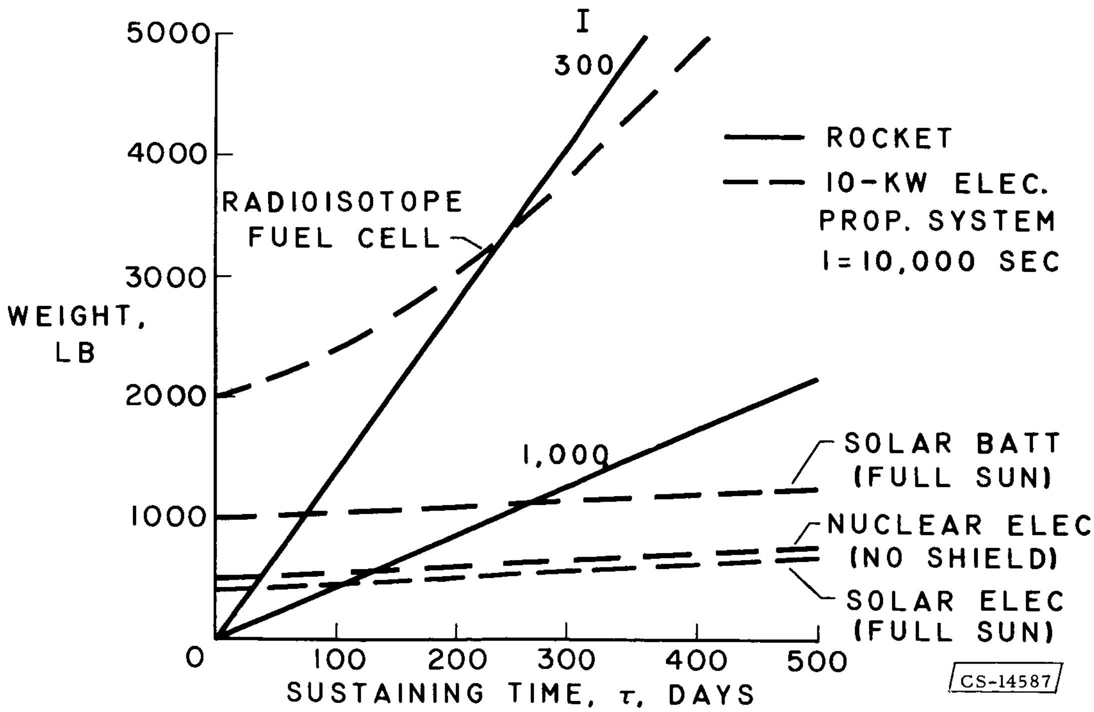

Figure 28. - Propellant plus powerplant weight for continuous thrust of 0.05 pound.

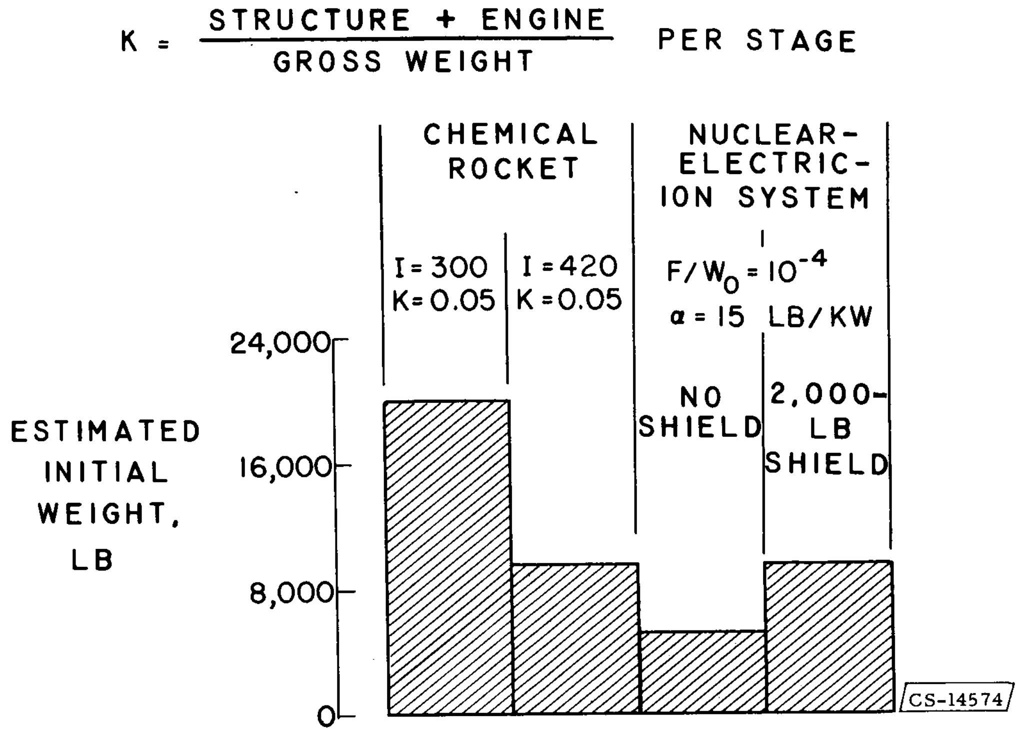

Figure 29. - Unmanned one-way Mars trip. Satellite-to-satellite; basic payload, 2000 pounds.

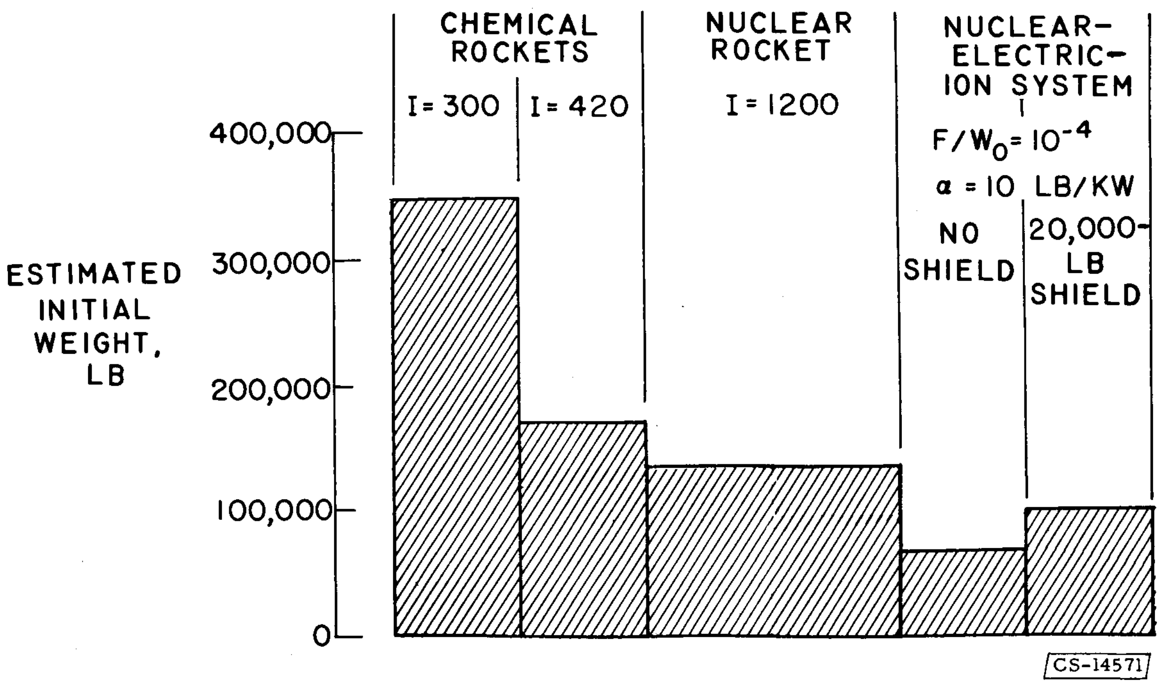

Figure 30. - Round trip to Moon. Eight-man crew; basic payload, 10,000 pounds; landing and exploration equipment, 16,000 pounds.

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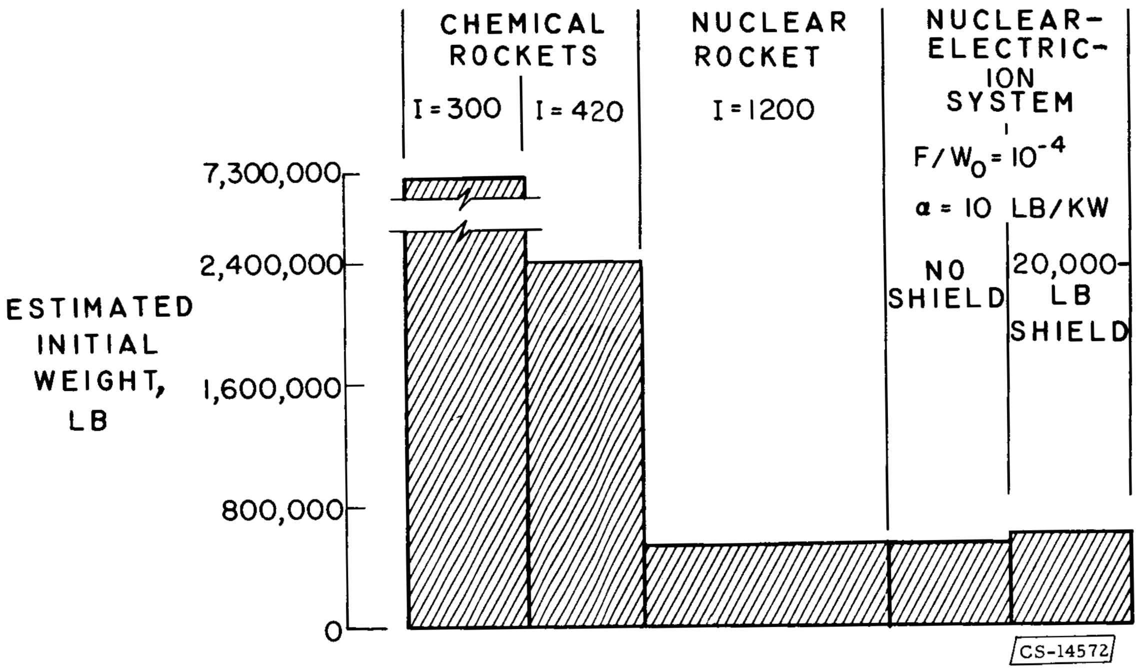

Figure 31. - Round-trip Mars expedition. Eight-man crew; basic payload, 50,000 pounds; landing and exploration equipment, 60,000 pounds.